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FINAL REPORT 7

SYSTEMS DEFINITION SUMMARY

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EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY (EOS)

PREPARED FOR:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

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CONTENTS

		Page
1.	INTRODUCTION	1-1
2.	STANDARD SPACECRAFT BUS	2-1
2.1	Performance Requirements	2-1
2.2	Modularity	2-1
2.3	External Interfaces	2-7
2.3.1	Launch Vehicle Interface	2-8
2.3.2	Shuttle Interfaces	2-11
2.3.3	Umbilical Connections	2-12
2.3.4	Observatory-to-STDN Interfaces	2-13
2.3.5	Observatory-to-TDRS Interface	2-13
2.3.6	Data Collection System Interface	2-15
2.4	Internal Interfaces	2-15
2.4.1	Internal Mechanical Interfaces	2-15
2.4.2	Thermal Interfaces	2-16
2.4.3	Electrical Power Interface	2-18
2.4.4	Data Interfaces	2-19
2.4.5	Safe Mode Bus	2-20
2.4.6	Spacecraft Harness	2-21
2.4.7	Electromagnetic Compatibility	2-22
2.4.8	Contamination Control	2-22
2.5	Redundancy and Reliability	2-24
2.6	Communications and Data Handling Module Design	2-26
2.6.1	Communications Equipment	2-30
2.6.2	Data Handling Equipment	2-34
2.7	Attitude Determination Module Design	2-35
2.8	Actuation Module Design	2-38
2.9	Power Module	2-42
2.10	Solar Array and Drive Module	2-45
2.11	Spacecraft Structure	2-47
3.	PAYLOAD DATA HANDLING	3-1
3.1	On-Board Data Formatting	3-1
3.2	Payload Data Transmission	3-4
3.2.1	Transmission to STDN Stations	3-5
3.2.2	Transmission to Local User Stations	3-6
3.2.3	Transmission to TDRS	3-7
3.3	STDN Reception of Payload Data	3-10

CONTENTS (Continued)

		Page
3.4	Central Data Processing Facility	3-11
3.4.1	Data Preparation	3-14
3.4.2	Data Refinement	3-16
3.4.3	Output Product Generation	3-17
3.4.4	System Management	3-18
3.5	Low-Cost Ground Stations	3-19
3.5.1	Acquisition Equipment	3-20
3.5.2	Direct Display	3-20
3.5.3	Record and Process	3-21
4.	APPLICATION OF THE MODULAR DESIGN TO VARIOUS MISSIONS	4-1
4.1	The EOS-A Mission	4-1
4.1.1	Payload Characteristics	4-1
4.1.2	On-Board Payload Data Handling	4-6
4.1.3	Standard Spacecraft Bus	4-7
4.1.4	Ground Data Handling	4-8
4.2	The EOS-B Mission	4-8
4.2.1	Payload Characteristics	4-10
4.2.2	On-Board Payload Data Handling	4-12
4.2.3	Standard Spacecraft Bus	4-12
4.2.4	Ground Data Handling	4-12
4.3	Solar Maximum Mission (SMM)	4-13
4.3.1	Solar Maximum Mission Payload	4-14
4.3.2	On-Board Payload Data Handling	4-14
4.3.3	Standard Spacecraft Bus	4-14
4.3.4	Ground Data Handling	4-16
5.	VERIFICATION CONCEPT	5-1
5.1	Qualification	5-2
5.2	Acceptance	5-4
5.3	Designing for Low-Cost Tests	5-4
5.3.1	Testability	5-4
5.3.2	Accessibility	5-8
5.3.3	Design Margins	5-8
6.	LOW COST MANAGEMENT APPROACHES	6-1

ILLUSTRATIONS

		Page
2-1	Standard Spacecraft Bus	2-3
2-2	System Maintenance Modes	2-5
2-3	Modular Spacecraft Block Diagram	2-8
2-4	Interfaces of Typical Observatory Using Standard Spacecraft Bus	2-9
2-5	Observatory/Launch Vehicle Mating Concepts	2-10
2-6	Module Attachment Interface Arrangement	2-17
2-7	Module Power Interface	2-18
2-8	Typical Module Power Interface	2-19
2-9	Data Handling Concept	2-20
2-10	Bus Formats	2-21
2-11	Communications and Data Handling Module	2-26
2-12	Communications and Data Handling Module Baseline Data	2-29
2-13	Data Handling Equipment Block Diagram	2-33
2-14	Attitude Control Functional Block Diagram	2-36
2-15	Attitude Determination Module Block Diagram	2-37
2-16	Attitude Determination Module	2-37
2-17	Star Tracker Geometry	2-38
2-18	Actuation Module (Piping not Shown)	2-39
2-19	Actuation Module (Minimum Redundancy)	2-41
2-20	The Power Module Can Accommodate up to Four Batteries; The Baseline Spacecraft Uses a 2-Battery Configuration	2-43
2-21	Power Module and Interfaces	2-43
2-22	Solar Array and Drive Module Block Diagram	2-46
2-23	Solar Array and Drive Module	2-46
2-24	Configuration for Module Structures for Spacecraft Bus	2-49
2-25	Structural Configuration of Standard Bus	2-50
3-1	MODS Multiplexer System Organization	3-3
3-2	Speed Buffer (LCGS) Partioning (Includes a Line Stripper and Formatting)	3-5
3-3	Wideband Communications Module	3-6
3-4	TDRS Antenna	3-9
3-5	Central Data Processing Facility Functional Structure	3-12

ILLUSTRATIONS (Continued)

		Page
3-6	Baseline Functional Image Processing System Design	3-13
3-7	LCGS Functional Block Diagram	3-19
3-8	LCGS Minimal Cost Record and Process Configuration	3-21
4-1	EOS-A with TDRS Antenna	4-4
4-2	Thematic Mapper Scan Geometry	4-4
4-3	Thor-Delta Configuration for EOS-B	4-9
4-4	HRPI Electronics Block Diagram	4-11
4-5	Solar Maximum Spacecraft	4-13
5-1	Qualification Model Integration and Test Flow	5-5
5-2	Flight Integration Test and Launch Operations Flow	5-6

TABLES

Table		Page
2-1	Standard Spacecraft Bus Candidate Mission	2-2
2-2	Redundancy Configurations	2-25
2-3	Properties of Redundancy Configurations	2-25
2-4	CDH Capabilities	2-27
2-5	Post-EOS-A Missions and Design Impacts	2-31
2-6	Battery Characteristics for Baseline Spacecraft	2-44
3-1	Payload Data Handling Requirements	3-2
3-2	Ku-Band Link to TDRSS-RF Link Calculation	3-8
4-1	Standard Bus Applicability to Selected Missions	4-2
4-2	Weight Summary	4-3
4-3	Thematic Mapper Characteristics	4-5
4-4	ERTS MSS Characteristics	4-6
4-5	Typical Instrument Complement for MSS	4-15
5-1	Overview of Recommended Versus Conventional Test Programs	5-1

GLOSSARY

ADM	Attitude Determination Module
APT	Automatic Picture Transmission
CCT	Computer Compatible Tape
CDH	Communications and Data Handling
CDPF	Central Data Processing Facility
CDIF	Cost Plus Incentive Fee
DCS	Data Collection System
DIU	Data Interface Unit
EGSE	Electrical Ground Support Equipment
EMC	Electromagnetic Interference
EMI	Electromagnetic Interference
EOS	Earth Observatory Satellite
ERTS	Earth Resources Technology Satellite
FSS	Flight Support System
GCP	Ground Control Point
GCR	Group Code Recording
GSE	Ground Support Equipment
HDDT	High-Density Digital Tape
HDMR	High-Density Multitrack Recorder
HRPI	High-Resolution Pointable Imager
LBR	Laser Beam Recorder
LCGS	Low-Cost Ground Station
MGSE	Mechanical Ground Support Equipment
MMD	Mean-Mission Duration
MODS	Multimegabit Operational Data System
MSS	Multispectral Scanner
NRZ	Non-return to Zero
NTTF	NASA Test and Training Facility
OGO	Orbiting Geophysical Observatory
OSO	Orbiting Solar Observatory
PE	Phase Encoded
PROM	Programmable Read-only Memory
RCP	Registration Control Point

GLOSSARY (Continued)

RF	Radio Frequency
ROM	Read-only Memory
SAMS	Shuttle-Attached Manipulator System
SMM	Solar Maximum Mission
SNR	Signal-to-Noise Ratio
SPMS	Specal-Purpose Manipulator System
STDN	Space Tracking and Data Network
TDRS	Tracking and Data Relay Satellite
TELOPS	Telemetry Operational System
TM	Thematic Mapper
TWT	Travelling Wave Tube
USB	Unified S-Band

1. INTRODUCTION

Accomplishments in space and advances in space technology over the past 15 years have been dramatic. Advancement of scientific knowledge has been a key goal, but the economic value gained from communication satellites, weather satellites, and more recently multispectral observations of the earth from space have provided a new justification for the space program.

In the present era of tight budgets, we must obtain the maximum benefit from each dollar spent, which strongly suggests that we must find ways to apply the hardware as well as the knowledge gained on one program to later programs, and ensure that new tasks are never allowed to become more difficult than absolutely necessary. In other words, the major emphasis in new programs must be on payload development and data use; supporting equipment must rely almost exclusively on standard designs. All urges to alter designs just because newer techniques are available must be strongly resisted. Cost-effectiveness over the range of anticipated missions, not optimization of supporting equipment performance for a specific mission, must dominate design. Compatible interfaces must be established and controlled at a high enough level to minimize redesign between missions and at a low enough level for multimission flexibility. These interfaces must permit module designs to exploit existing equipment and rarely force advances in the state of the art.

NASA has faced these issues wisely and has undertaken a program to define a standard spacecraft bus for performing a variety of earth orbiting missions of the late 1970's and 1980's. This report summarizes work done by TRW under contract NAS5-20519 in support of this effort at GSFC. Our work has focused on low-cost, multimission capability, benefiting from the Space Shuttle System when it becomes available, and making use of conventional launch vehicles for the near-term.

We have reviewed work on EOS undertaken at GSFC during the past several years, recommended improvements indicated by our analyses, and otherwise verified GSFC concepts. Our study results have been detailed in Reports 1 through 6 issued earlier. In summary, we have reached the following general conclusions:

- There is sufficient commonality in support requirements across a range of missions and a broad enough technology base to justify the use of a standard spacecraft bus for medium-weight (1500 to 6000 pounds) earth-orbiting missions.

- A modular bus employing well thought out and rigidly enforced interface specifications can lead to cost reductions over a conventional design even on the first procurement, further reductions when the design and fabrication costs are amortized over its multimission usage, and dramatic reductions when coupled with the Space Shuttle System's ability to perform on-orbit maintenance by module-for-module substitution.

- Modularity is implemented best by incorporating communication and data hanling functions into two modules, one for normal housekeeping and the other for payload data output; incorporating power functions into two modules, one for energy storage and bus regulation, and the other for the solar array and its drive; and incorporating attitude and orbit control functions into two modules, one for attitude determination and the other for attitude and orbit actuation.

- An on-board computer facilitates such functions as instrument and antenna pointing, electric power management, thermal control, attitude stabilization, and on-board health monitoring with precision and simplicity either otherwise unimplementable or implementable at excessive cost, complexibility, and/or unreliability. Many mission-to-mission changes can be accomplished easily in software with minimum or no hardware changes.

- Modularity can and should be extended to the ground data handling. Standard receivers, data synchronizers, tape recorders, formatting and correcting functions, and archived media can be operated at a wide variety of data rates and formats with little or no change and at a wide variety of data quantities through incremental changes in equipment numbers. Special-processing functions (e. g., synthetic-aperture radar beam synthesis) will always employ special-processing equipment in addition to the standard equipment.

- A standard, modular spacecraft bus, a modular ground data handling system, and improved resolution multispectral imaging instruments can yield a series of meaningful EOS missions, extending earth observation techniques and providing a sound technological basis for future operational systems.

- Innovative management techniques requiring changes on the part of both contractor and government can reduce costs significantly.

These involve a longer range commitment on both sides, greater attention to the commonality of the motivations and need for protection of contractors and the government, penalties on both sides for initiating changes, and closer coordination between government and contractor to ensure that goals are realistic and impending difficulties are quickly identified and resolved.

The following section presents our standard spacecraft bus illustrating its design, how modularity is achieved, and the range of capabilities applicable to future missions. We then present our ground data handling design in its most general form and indicate how it might be applied to missions requiring a variety of throughput capabilities, processing functions, and output-product formats. Next, we show how these designs can be combined with several payloads to yield systems for performing various missions and how a new integration and test approach can provide needed confidence at reduced cost. Finally, we present our thoughts on low-cost management.

2. STANDARD SPACECRAFT BUS

The standard spacecraft bus is the equipment complement needed to support a payload, excepting payload-peculiar communications and data handling equipment and the structure that supports them. (See Figure 2-1.) Such support capabilities as power, attitude, weight accommodation, telemetry, and commands can be varied over the range of requirements without changing the spacecraft design. The standard bus is modular; it allows on-orbit servicing by Shuttle and simplifies the synthesis of future missions, minimizing the number of design decisions needed to achieve spacecraft/payload compatibility.

The bus with its associated payload can be launched by Shuttle or by any of the following expendable launch vehicles:

- Thor-Delta 2910 or 3910
- Titan III B, C, or D.

Launches may be from ETR or WTR, depending on the final orbit inclination specified. Orbit altitudes may range from 200 to 1000 n mi at near arbitrary inclinations and at or near geosynchronous altitude at inclinations in the range of 0 to 28 degrees.

The observatory can accommodate either retrieval or on-orbit module replacement by the Shuttle for low-earth orbit missions. Each module is structurally independent and removable from the spacecraft by the Shuttle special-purpose manipulator system (SPMS).

2.1 PERFORMANCE REQUIREMENTS

Table 2-1 presents a series of candidate missions, which could benefit from a low-cost standard spacecraft bus, and values for the most significant performance parameters. From these the required range of values for each parameter can be abstracted as a guide to the spacecraft design.

2.2 MODULARITY

Modularity concepts for the standard spacecraft bus were evolved based on these guidelines:

Table 2-1. Standard Spacecraft Bus Candidate Missions

MISSION	MISSION DESCRIPTION	TYPICAL PAYLOAD INSTRUMENTS[1]	PAYLOAD CHARACTERISTICS AND REQUIREMENTS[1], [2]			DATA RATES	ATTITUDE CONTROL[2]		ORBIT AND LAUNCH VEHICLE
			WEIGHT (LB)	VOLUME (CU. FT)	POWER WATTS		ORIENTATION	PERFORMANCE	
EOS A	LAND RESOURCES MANAGEMENT SENSOR DEVELOPMENT	• THEMATIC MAPPER • 5-BAND MSS	430	51	190	135 MBIT/SEC	EARTH-POINTING	30 ARC SEC ACCURACY	705 KM SUN SYNCHRONOUS, DELTA 2910 LAUNCH
EOS B	LAND RESOURCES MANAGEMENT SENSOR DEVELOPMENT	• THEMATIC MAPPER • HRPI	573	70	222	240 MBIT/SEC	EARTH-POINTING	30 ARC SEC ACCURACY	705 KM SUN SYNCHRONOUS, DELTA 3910 LAUNCH
EOS C	EXPERIMENTAL OCEANOGRAPHY AND METEOROLOGY	• ADVANCED ATMOSPHERIC SOUNDER • OCEAN SCANNING SPECTRO PHOTOMETER • SEA SURFACE TEMPERATURE IMAGING RADIOMETER • PASSIVE MICROWAVE RADIOMETER • CLOUD PHYSICS RADIOMETER	500	12	450	1 MBIT/SEC	EARTH-POINTING	0.25 DEGREE ACCURACY	700 NMI SUN SYNCHRONOUS, SHUTTLE LAUNCH
SEASAT	DEMONSTRATE SPACE MONITORING OF OCEAN SURFACE CONDITIONS	• SYNTHETIC APERTURE RADAR • PASSIVE MICROWAVE RADIOMETER • INFRARED IMAGER • DATA COLLECTION SYSTEM	500	600	500	0.5 KBIT/SEC TO 10 MBIT/SEC	EARTH-POINTING	0.25 DEG ACCURACY	LOW ALTITUDE, NON-SUN SYNCHRONOUS ORBIT (BASELINE IS 391 N MI, 82 DEG INCLINED); THOR-DELTA LAUNCH FOR THIS PAYLOAD
SOLAR MAXIMUM MISSION (SMM)	STUDY FUNDAMENTAL MECHANISM AND EFFECTS OF SOLAR FLARES	• ULTRAVIOLET MAGNETOGRAPH • EUV SPECTROMETER • X-RAY SPECTROMETER • HARD X-RAY IMAGER • LOW-ENERGY POLARIMETER	1430	13.5	175	5 KBITS/SEC	SUN-POINTING	5 ARC-SEC ACCURACY; 1 ARC-SEC DRIFT IN 5 MINUTES	300 N MI, 33 DEG INCLUDING ORBIT; THOR-DELTA LAUNCH
SYNCHRONOUS EARTH OBSERVATORY SATELLITE (SEOS)	RESOURCE AND WEATHER MONITORING FROM STATIONARY PLATFORM; TIMELY WARNINGS AND ALERTS	• LARGE APERTURE SURVEY TELESCOPE • MICROWAVE SOUNDER • FRAMING CAMERA • ATMOSPHERIC SOUNDER AND RADIOMETER	2640	350	145	60 MBIT/SEC	EARTH-POINTING WITH SCAN	POINT TO 5 ARC-SEC (1σ) STABILITY; 1 ARC-SEC (1σ) IN 12 MINUTES	24-HOUR GEOSTATIONARY; LATITUDE AND LONGITUDE STATIONKEEPING; LAUNCHED ON SHUTTLE OR LARGE CONVENTIONAL LAUNCH VEHICLE WITH "TUG" STAGE

NOTES: (1) DATA FROM THE FOLLOWING REPORTS:
- "SEASAT-A PHASE I STUDY REPORT," W. E. SCULL, NASA/GSFC, AUGUST 1973.
- "SOLAR MAXIMUM MISSION (SMM) CONCEPTUAL STUDY REPORT," NASA/GSFC REPORT X-703-74-42, JANUARY 1974.
- "SYNCHRONOUS EARTH OBSERVATION SATELLITE (SEOS)," NASA/GSFC DOCUMENT, 1974.

(2) DATA FROM "SMM, SEASAT, ERS AND SEOS INSTRUMENT TABLES," NASA/GSFC, 1974.

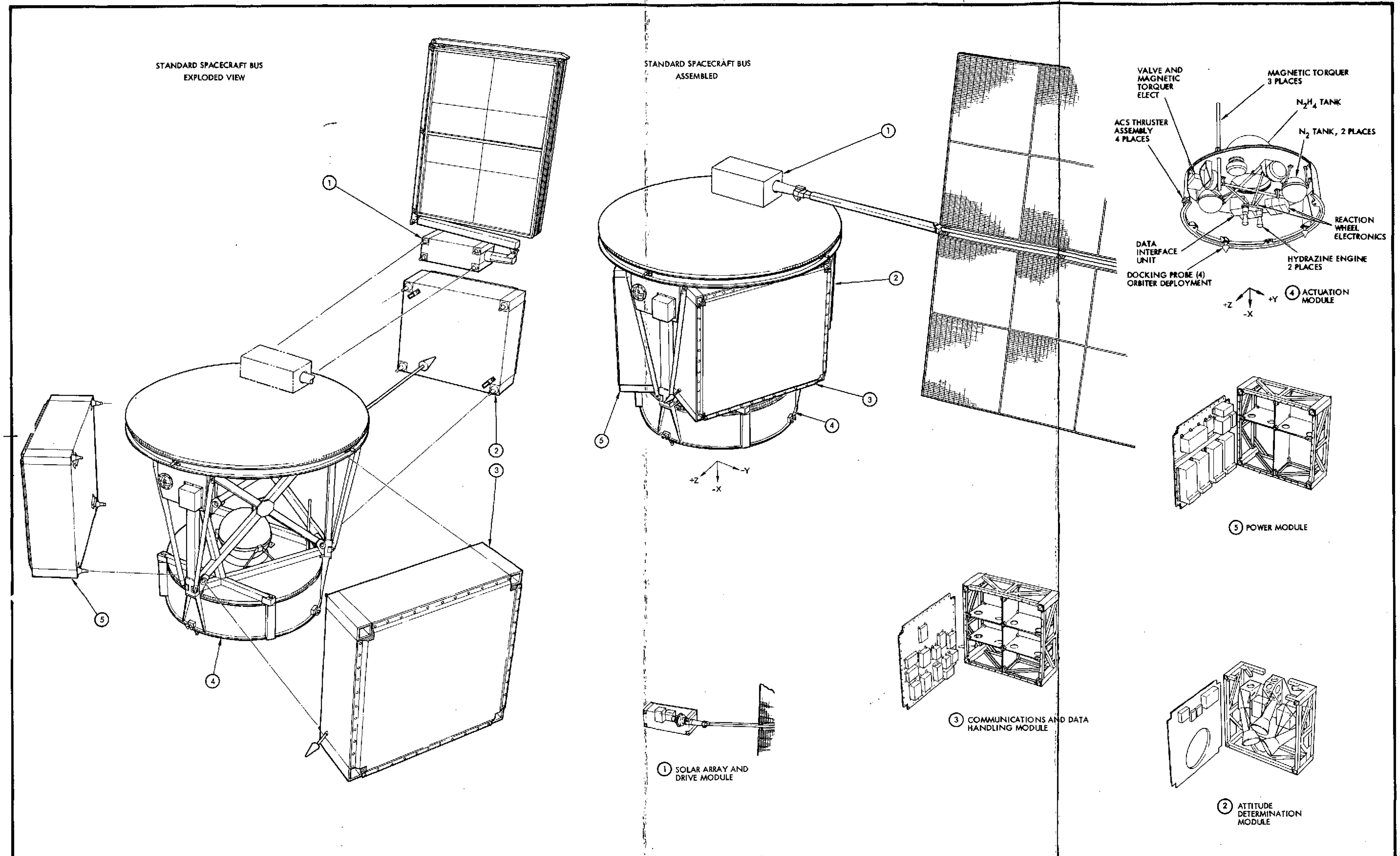

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Figure 2-1. Standard Spacecraft Bus

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- Reinforce overall technical feasibility and cost-effectiveness by applying existing technology and hardware to the maximum.
- In configuring the spacecraft subsystems and payload interface, allow for substantial growth and flexibility.
- Minimize recurring costs of hardware by eliminating highly optimized designs, excessively tight tolerances, and exotic materials.
- Group mission-independent functions into logical self-contained modules that can satisfy a range of missions. Group mission-independent functions into payload modules that are easy to alter from mission-to-mission.
- Incorporate sufficient design margin to minimize testing without sacrificing confidence in performance over the mission lifetime.

Studies and tradeoffs have shown that in-orbit servicing is the most economical among the three possible modes of spacecraft maintenance (Figure 2-2). The modular concept provides this capability. Modularizing subsystem equipment as far as practicable and producing many units in a single run reduces costs substantially. Modules usable for many diverse missions with minimum change satisfy NASA's low-cost spacecraft goals.

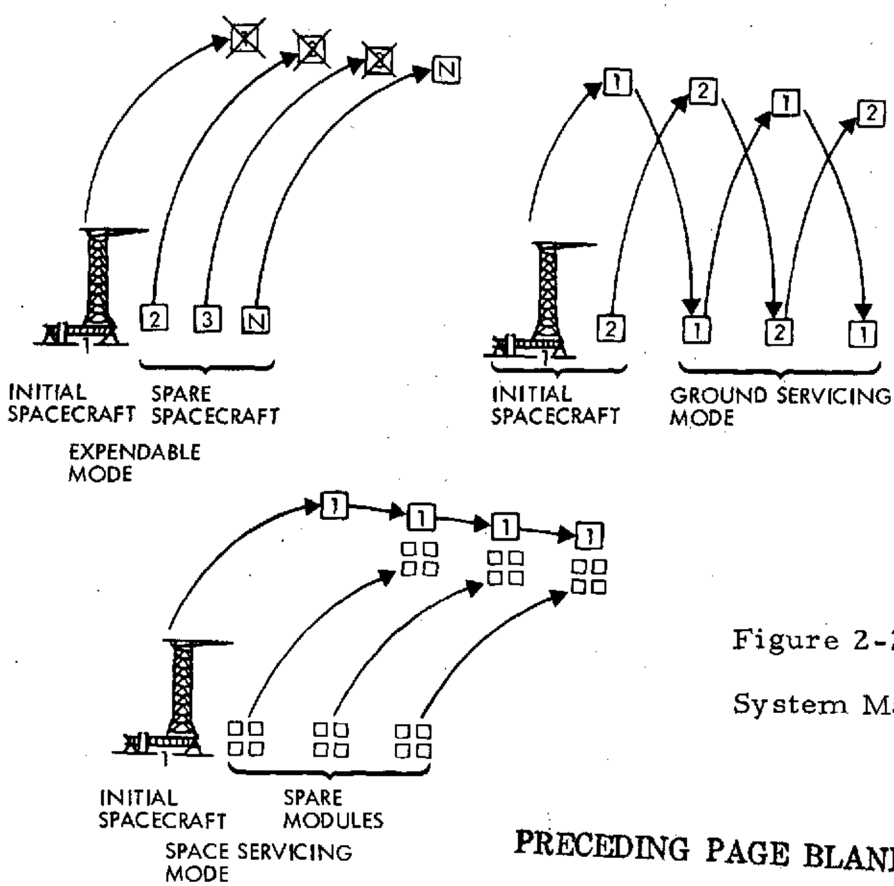

Figure 2-2

System Maintenance Modes

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The standard spacecraft bus has five modules:

1) Electric power module, containing the power control unit and sufficient batteries to support the mission payload.

2) Communication and data handling module, with receivers, transmitters, omni antennas, and computer.

3) Attitude determination module, with star trackers, inertial reference unit, magnetometer, and sun sensor.

4) Solar array and drive module, containing array drive, slip rings, and drive electronics and supporting a mission-dependent solar array.

5) Actuation module, with momentum wheels, magnetic torquers, nitrogen reaction control system, and hydrazine orbit transfer and orbit maintenance system. The capacities and capabilities of the elements of this module are payload-dependent.

The first three are very similar structurally. The power module includes stiffening shear webs to support the outboard radiator panel and weighs more than the other two. The actuation module occupies a cylindrical volume at the aft end and the triangular volume within the spacecraft structure. The solar array and drive module consists of a support structure for the drive, slip rings and associated electronics, and the array structure into which an integral number of standard subpanels are installed. The solar array module is mounted above the transition ring; during launch the array is folded and attached to the payload support structure.

To arrive at a specific set of standard module dimensions, we considered spacecraft constraints imposed by Thor-Delta launches. The actuation module is payload-dependent and its volume is variable. Module depth for the first three modules was determined by the size of the largest subsystem elements, batteries, and momentum wheels. Module width is the largest permitted when the modules are arranged in triangular fashion within the shroud dynamic envelope. Module length is then established to give sufficient room to contain all the equipment. A 48 x 48 x 18 inch size for the standard modules satisfies these requirements.

Figure 2-3 is the modular spacecraft block diagram. Some of the modularity features built into the standard spacecraft bus and payload modules are as follows:

- All structural, dynamic, thermal, and electrical interaction between spacecraft and payload modules is at the transition ring interface.
- Payload and spacecraft module mechanical and electrical interfaces are standardized.
- Spacecraft and payload modules are thermally decoupled from the spacecraft and payload structures. Thermal interfaces are the same for both nonrefurbishable and in-orbit serviceable missions.
- Modules can be replaced in orbit; module thermal, electrical, and mechanical designs are compatible with the Shuttle module exchange mechanism.
- For non-space refurbishable missions, module replacement mechanisms are deleted allowing payload weight. Separate payload module structures, which provide the interface with the SPMS for in-orbit exchange, are deleted to save weight.
- Modules have substantial design margins and volume for growth and flexibility to accommodate a wide range of payloads and missions.
- Spacecraft and payload modules are fed electrical power on two independently controlled main buses; structure and module heaters are fed power on an independentally controlled bus.
- Each module receives commands and communications on its own party line via a data bus that is physically separated from high-level buses.

2.3 EXTERNAL INTERFACES

A key factor in modularizing the spacecraft design and making major cost reductions is the definition and control of interfaces, both internal to the spacecraft and between the spacecraft, the payload, the launch vehicle, and ground data handling. To fulfill program goals, realistic and compatible interface specifications are essential. Any specification that forces an advance in the state of the art is not realistic and defeats GSFC's low-cost goals. To provide compatibility, margin must exist between the tightness of control of a parameter at its source and the tolerance to changes in the parameter at its destination. This

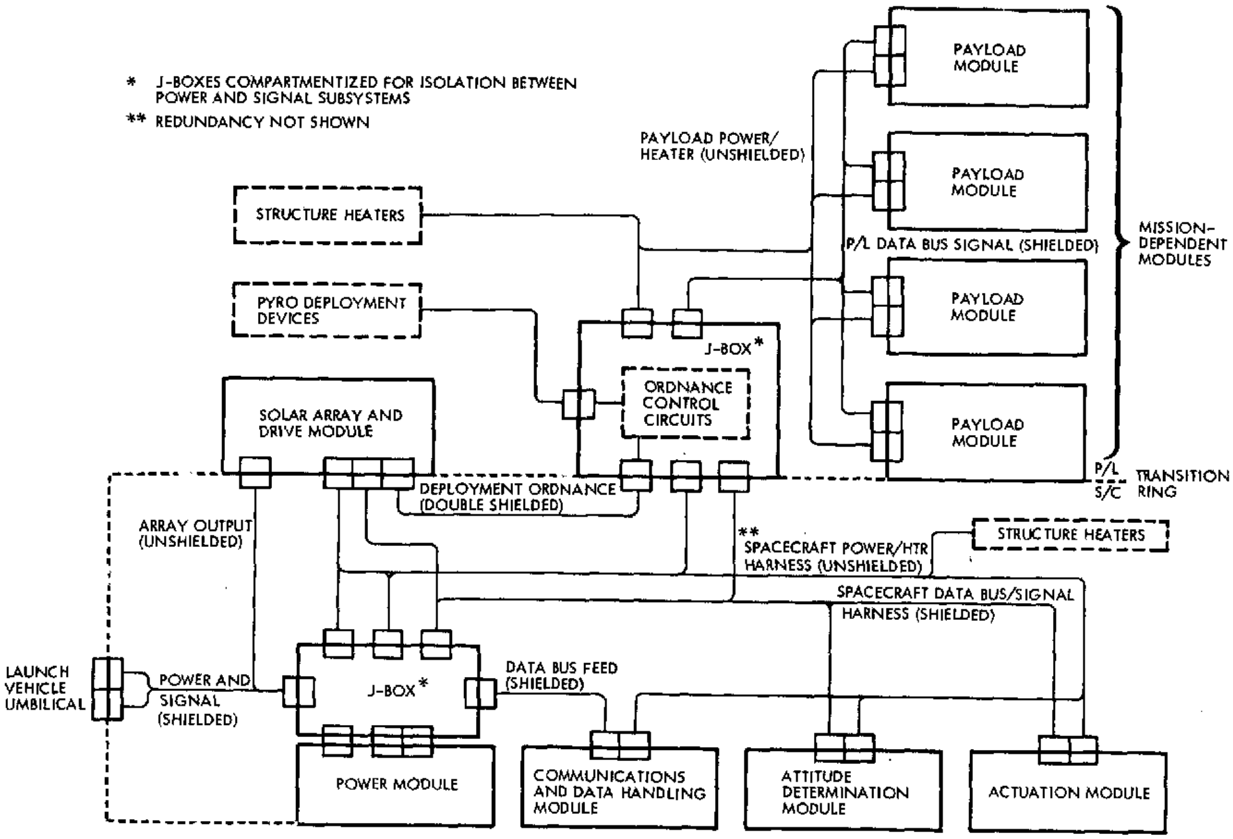


Figure 2-3. Modular Spacecraft Block Diagram

approach ensures that system-level performance can be met through module-level tests, a cost-reducing feature that is mandatory if on-orbit servicing is to be practical. Once established, interface specifications must be rigidly enforced; all attempts to alter them to ease a particular design problem must be resisted.

Figure 2-4 shows the internal and external interfaces of a typical observatory using the standard spacecraft bus. This section discusses the external interfaces. Internal interfaces are covered in Section 2. 4.

2. 3. 1 Launch Vehicle Interface

The standard spacecraft bus can be launched by Thor-Delta 2910, 3910; Titan IIIB, C, D; Titan IIIE/Centaur; and Space Shuttle (see Figure 2-5). An observatory/launch vehicle adapter normally mates with and supports the observatory at the transition ring during launch and retrieval operations (cases 2 and 3, Figure 2-5). For some low payload-weight missions employing expendable boosters, the adapter may be attached aft of the spacecraft structure (case 1, Figure 2-5), reducing

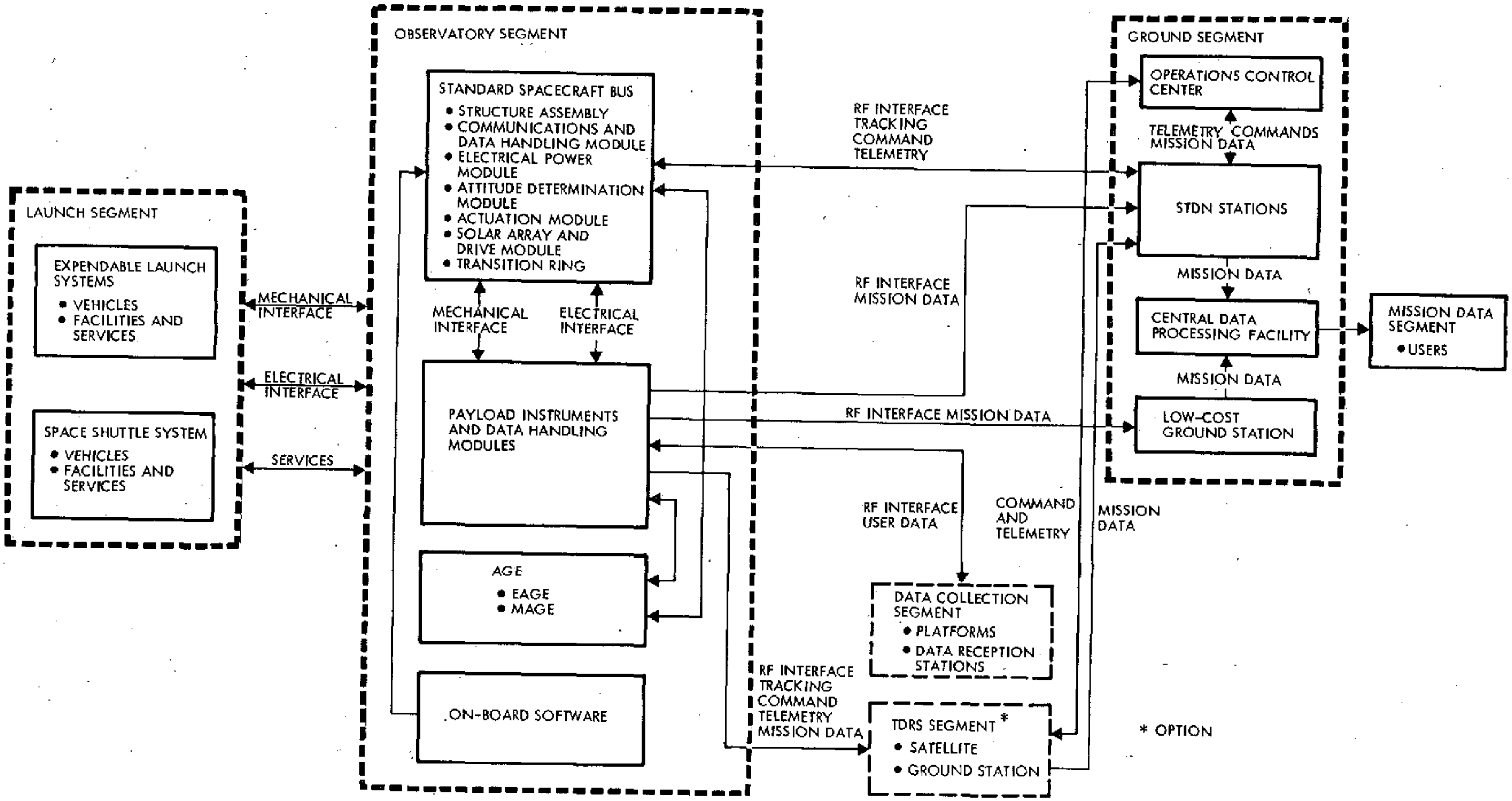


Figure 2-4. Interfaces of Typical Observatory Using Standard Spacecraft Bus

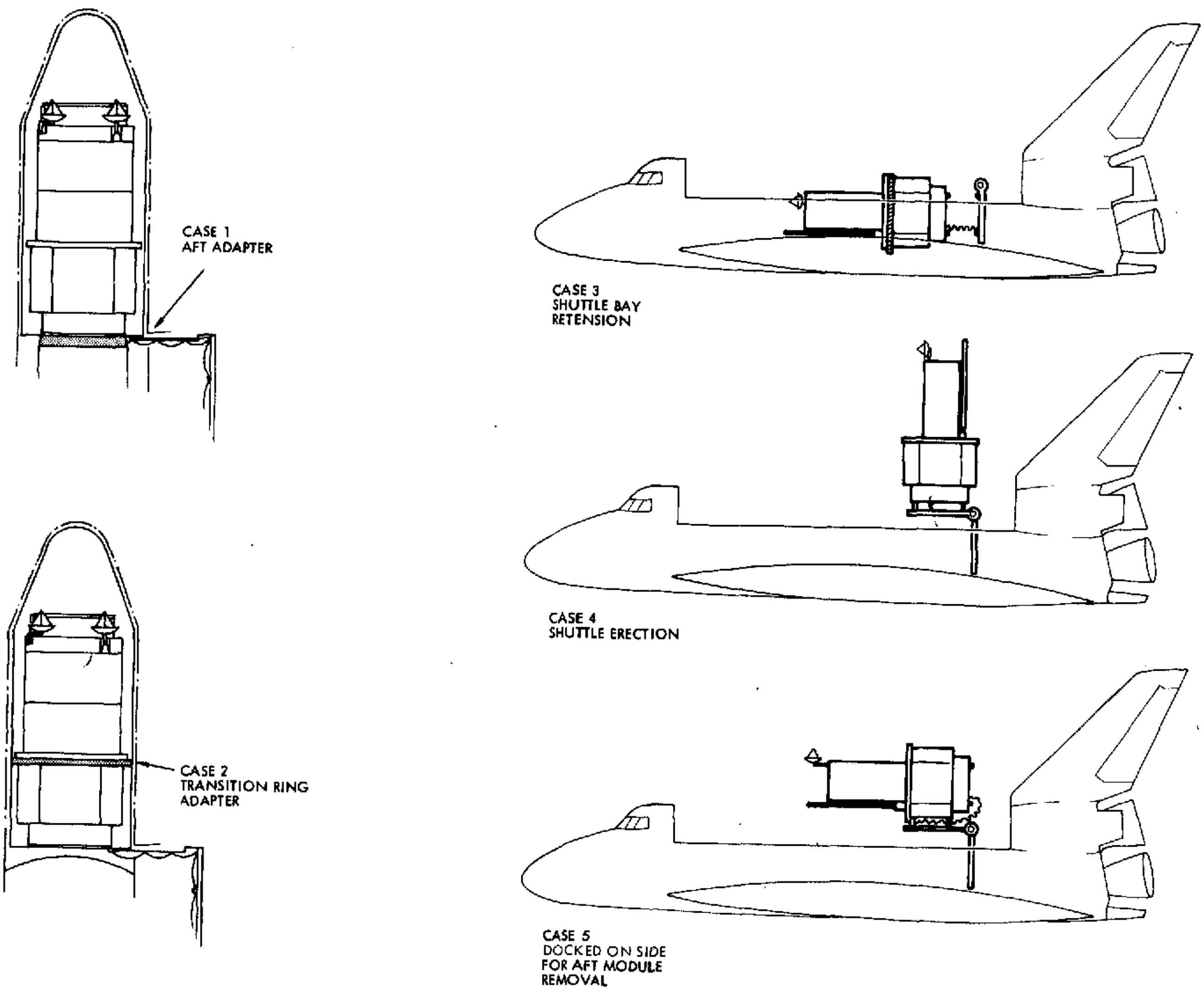


Figure 2-5. Observatory/Launch Vehicle Mating Concepts

adapter weight and adding, for the Thor-Delta vehicle, 280 pounds of payload capability.

The spacecraft structure, strong enough to withstand total observatory loads on Delta launches with an aft adapter, also can accommodate missions requiring heavier actuation modules, including more on-board propulsion, with the transition ring as the primary load path. Thus, the same spacecraft structure can accommodate a wide variety of payload weights and can be launched from Thor-Delta, Shuttle, and Titan without redesign.

2.3.2 Shuttle Interfaces

The Shuttle-orbiter interfaces with the observatory for launches, on-orbit servicing (module exchange), and return of the entire observatory to earth (retrieval).

The basic mechanical interfaces between EOS and Shuttle are provided by the flight support system (FSS), which consists of:

- Transition ring cradle to support the observatory during launch and return to earth
- Docking adapter to erect and to hold the observatory during module exchange
- Special-purpose manipulator system (SPMS) which performs the module exchange.

In addition, the Shuttle-attached manipulator system (SAMS), part of the standard orbiter equipment, maneuvers the observatory in and out of the docking adapter.

The mechanical interface between the observatory and the docking adapter consists of four drogue probes hard mounted to the spacecraft aft structure. The docking adapter contains the clasping mechanisms, which are normalized to the observatory temperature prior to docking. A drogue-to-adapter socket is used to position the umbilical connection for mating. The docking adapter erects the spacecraft to a vertical position and rotates it to make the modules accessible.

SAMS is used for initial grappling of the observatory upon rendezvous and for positioning it on the docking platform. SAMS also is used to retrieve certain modules (i. e., solar array) from forward stowage positions in the Shuttle bay. SAMS may also be used to deploy and collapse appendages, such as antennas and the solar array.

The observatory is supported and held within the Shuttle bay to the transition ring cradle of the FSS via three pin-like load fittings at the transition ring station. These fittings are mounted to the transition ring, which in turn is used as a grappling hold by the SAMS for observatory handling. The FSS transition ring cradle has compatible mechanisms to support the observatory load fittings. Compared to complete encirculation of a transition ring by a clam-shell type clamp, the three-point suspension provides a simpler FSS cradle. In this way the transition ring need not be elevated to clear the clamp and there are no critical thermal distortion and fabrication tolerances between ring and clamp.

2.3.3 Umbilical Connections

At certain times during the mission, the observatory must be coupled by an umbilical cable for telemetry power and exchange:

- Prior to launch: aft adapter, conventional launch vehicle
- Prior to launch: transition ring adapter, conventional launch vehicle
- Shuttle bay: while retained in the flight position
- Erected for module exchange while docked to Shuttle
- Docked to Shuttle on side while exchanging aft module.

These are accomplished with a single umbilical connector at the aft end of the observatory structure. For prelaunch operations with Delta or Titan, the umbilical is released by a mechanism from the launch gantry. A door is provided in the fairing for this purpose. During launch, the observatory is autonomous.

For Shuttle launch or retrieval, the observatory is secured in the bay by its transition ring and the umbilical comes from the docking adapter of the FSS. A mechanism within the docking adapter plugs and releases the umbilical when EOS is docked. The umbilical connector requires a mechanical latch for positive engagement and retention of the observatory in the Shuttle bay. (The docking adapter can back away 2 inches while the umbilical remains attached.)

During aft end access for actuation module removal, while the observatory rests docked on its side by means of auxiliary docking drogues on one of the spacecraft modules, the umbilical requires a few feet of service loop (as opposed to a few inches for the other cases).

The umbilical has about 50 pin connectors and is a conventional design except for the positive mechanical latch.

2.3.4 Observatory-to-STDN Interfaces

The observatory interfaces with the STDN stations and is compatible with the unified S-band system for tracking, command, and telemetry (transmission of payload data is covered in Section 3.2.1). The RF links between the observatory and STDN stations have these characteristics:

Uplink (USB)

Frequency:	2050 to 2150 MHz range
Data rate:	2000 bit/sec
Error rate:	$< 1 \times 10^{-10}$
Antenna polarization:	Singly linearly polarized or dual right and left circular
Command modulation:	PCM/PSK/FM/PM

Downlink (USB)

Frequency:	2200 to 2300 MHz range
Narrowband telemetry rate:	Selectable 64, 32, 16, 8, 4, 2, 1 kbit/sec
Narrowband modulation:	Split phase PCM/PMPSK on subcarrier
Medium-rate telemetry:	512 kbit/sec maximum
Medium-rate modulation:	Split phase PCM/PM

2.3.5 Observatory-to-TDRS Interface

For some missions, the TDRS system (when available) may be required to support the observatory by relaying tracking, telemetry, command, and payload data between the observatory and ground stations. This link permits data transmission and commanding at all times when the observatory is within line-of-sight of the TDRS. The observatory may also use TDRS range and range rate tracking data to support ephemeris determination.

Characteristics of the TDRS telemetry and command link interface with the observatory are as follows (transmission of payload data to TDRS is covered in Section 3.2.2).

Forward (Command Link)

Frequency:	2106.4 MHz (TDRSS multiple access) 2025 to 2120 MHZ (single access)
EIRP:	13 dBw (multiple access) 11.5 dBw (single access)
Bandwidth:	5 MHz (multiple access) 20 MHz (single access)
Command rate:	100 to 1000 bit/sec (multiple access)
Antenna gain:	23 dB (multiple access) 35.4 dB (single access)
Modulation:	PN spread spectrum
Chip rate:	0.6 MCPS (multiple access) 6.0 MCPS (single access, normal power) 10.0 MCPS (single access, high power)

Return (Telemetry Link)

Frequency:	2287.5 MHz (multiple access) 2200 to 2300 MHz (single access)
TDRSS antenna gain:	28 dB (multiple access) 36.0 dB (single access)
Bandwidth:	5 MHz (multiple access) 10 MHz (single access)
Maximum telemetry rate:	48 kbit/sec (multiple access) 5 Mbit/sec (single access)
Modulation:	PN spread spectrum (3 MCPS, multiple access) PSK (single access)
System noise temperature:	824°K (multiple access) 824°K (single access)

Observatory to TDRS Maximum Range:	42,800 km
Maximum Doppler Shift:	± 47.2 kHz (forward link) ± 51.2 kHz (return link)

2.3.6 Data Collection System Interface

A data collection system (DCS) interface may be required on some missions to relay ground instrument data from remote locations to ground collection stations. In this event, the observatory carries a DCS module that implements the RF relay between the remote location and the collection station when both are within line-of-sight with the observatory.

2.4 INTERNAL INTERFACES

To achieve a workable modular design concept, i.e., one in which modules can be built, tested, and expected to operate together without after-the-fact tailoring, each of the following interfaces among modules and between modules and structure must be established and controlled:

- Mechanical; including size, mounting, strength, and mass properties
- Thermal; including radiative and conductive exchange of thermal energy
- Power; including voltage level, regulation, and source and destination impedances
- Data; including transfer rates, synchronization, logic levels, formats and impedances
- Safe mode; including conditions for entering and returning
- Harness; including connectors and J boxes
- EMC; including emission and susceptance levels for both conducted and radiated energy
- Contamination; including emission and susceptance

Our approach to establishing and maintaining these interfaces is covered in the following paragraphs. The discussion is slanted towards the standard spacecraft bus but applies as well to the entire observatory.

2.4.1 Internal Mechanical Interfaces

Module attachment to the spacecraft structure involves the following design considerations:

- Launch load transfer and rigid body stiffness in orbit

- Thermal resistance to interaction between module and spacecraft structure
- Accountability for misalignment due to fabrication tolerances
- Removability from the observatory by the SPMS after installation of suitable mechanical attachment devices
- Alignment after attachment of critical elements, such as attitude determination and payload sensors
- Closure and separation forces for the electrical connector
- Simplicity and reliability of operation by the Shuttle SPMS and astronaut/operator
- Immunity from cold welding, corrosion, contamination, lubrication life, and outgassing and susceptibility to damage
- Minimum weight per attachment fitting
- Control of center of gravity when substituting modules

The design presented here minimizes load transfer through modules to preserve alignments and to reduce the thermal conductivity that would exist with a more rigid interface. The attachment interface consists of four identical probe-like mechanisms mounted to the rear of each module assembly. Each probe has a mating female socket attached to the spacecraft frame structure. By selective clearance, each of the four fittings is constrained in only those directions indicated in Figure 2-6. Each male probe has an attach bolt, which is turned by SPMS to a predetermined torque, mating the electrical connectors and securely bolting module to structure. The module is removed in a similar manner.

As presently envisioned, the insertion of the module into the spacecraft structure by SPMS requires astronaut/operator guidance during the final few inches of travel because of the tolerance buildup involved in all the systems. Without attach design, the SPMS is only required to grapple the module for transport and to rotate each latch. All modules, including the solar array, have identical attachment mechanisms, operable with either SPMS or SAMS with a special end effector.

2.4.2 Thermal Interfaces

Thermal interactions in most spacecraft designs cause substantial cost in design and system testing. This problem is bypassed in the

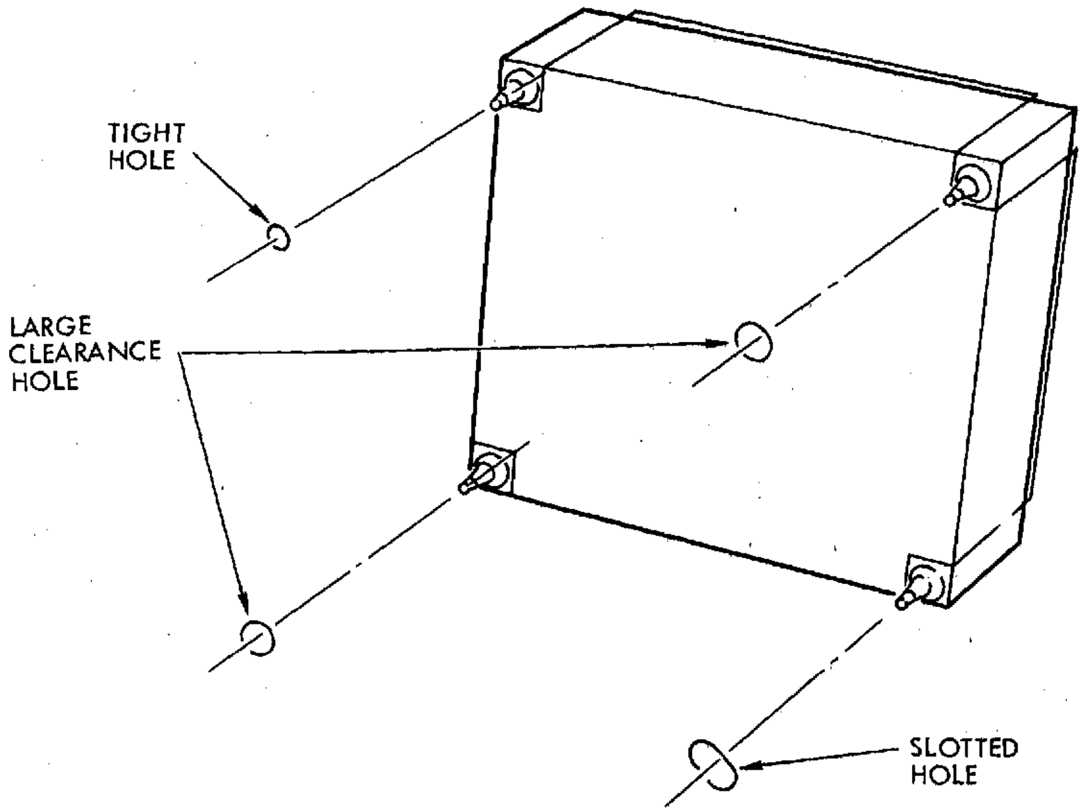


Figure 2-6. Module Attachment Interface Arrangement

standard spacecraft bus by thermally isolating each module. Five of the six sides of each module have thermal insulation; the remaining side, which sees no other observatory part, is provided with an area having a high α/ϵ surface to dissipate internally generated heat. Approximately 16 square feet of surface area is available in each module, but much of the radiating surface is masked off to achieve the desired equilibrium temperature. Thermal control is then maintained by internal heaters. All highly dissipative elements are mounted to the radiative surface to minimize thermal gradients.

Conductive heat flow at the points where each module attaches to the structure is less than 1 watt per attachment. The structure also is insulated and has heaters. Under computer control, structural members are held closely to the design temperature to minimize deformation that could degrade instrument pointing accuracy and to minimize thermal interaction with the modules.

2.4.3 Electrical Power Interface

The power subsystem design provides a standard interface for each observatory module. Fault isolation prevents failures in any module from affecting another. The prime power distributed to all modules is 28 ±7 volts. Maximum expected range for orbits below 900 nautical miles is 26 to 32.5 volts. The expected range for the geostationary orbit does not exceed 22 to 32.5 volts, assuming a possible 60 percent depth of discharge under normal operating conditions.

Prime power is isolated from secondary power in each module by a regulated converter, which provides ±5 percent regulation full load to 50 percent load. Individual regulators at the black box level provide closer tolerance on supply voltages where required. The primary power and heater bus interfaces to each module are shown in Figure 2-7. This design allows either of two buses to power the module equipment independent of the selected redundancy within the module.

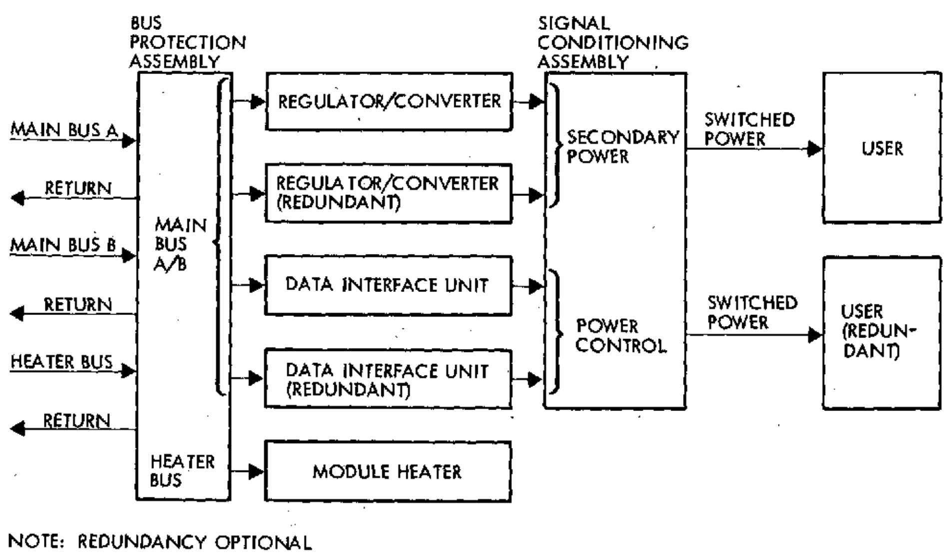


Figure 2-7. Module Power Interface

One converter powers all equipment within the module with one exception. Each data interface unit has a separate converter so that a single-point failure in the equipment cannot prevent commanding of the module. The primary power bus for load faults is isolated by fuses shown in Figure 2-8. The fuses are paralleled providing redundancy for a fuse

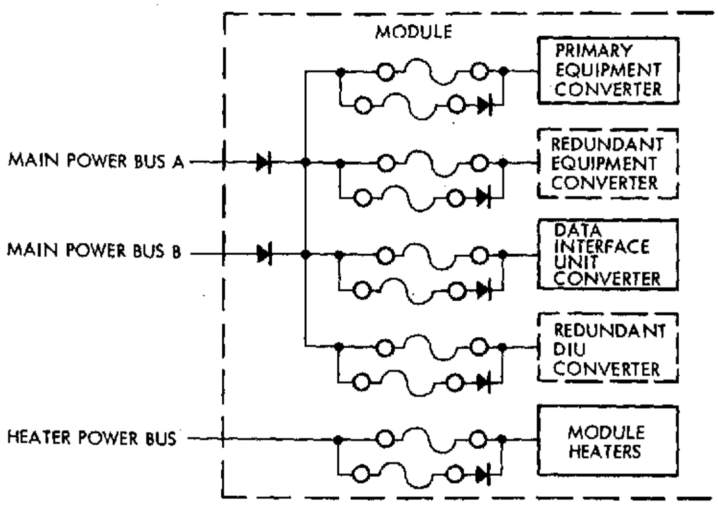


Figure 2-8

Typical Module Power Interface

opened in launch vibration. The diode in series with one fuse prevents current flow in that leg until the other fuse has opened. Each fuse may then be sized for a reasonable fault current to minimize the effect on the primary power bus. Use of the diode does not significantly increase the time required to open the line after a fault.

2.4.4 Data Interfaces

Data transfer between electrically isolated payload and spacecraft modules is handled by an AC-coupled party-line data bus (see Figure 2-9). This system has substantial flexibility to accommodate a range of missions. The bus controller has basic autonomy; telemetry data from the various data interface units (DIU) are collected in a standard format and commands received from the ground are distributed in real time. Its full capacity, however, comes into play under control of the on-board computer. In this mode a wide variety of telemetry formats is possible and commands can be executed after any predefined delay. Furthermore, data can be gathered and instructions issued to any DIU as part of a computer program, e.g., attitude control or temperature control of the structure.

The data bus system can interface with up to 32 modules and telemetry and command capacity of each module can be expanded by up to eight standard slices.

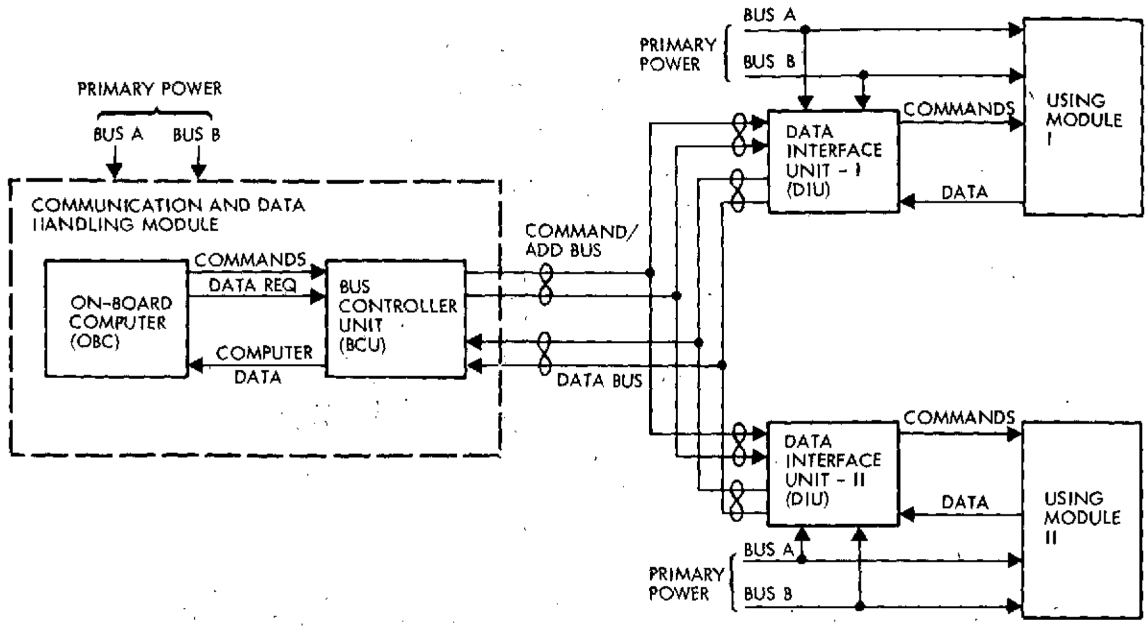


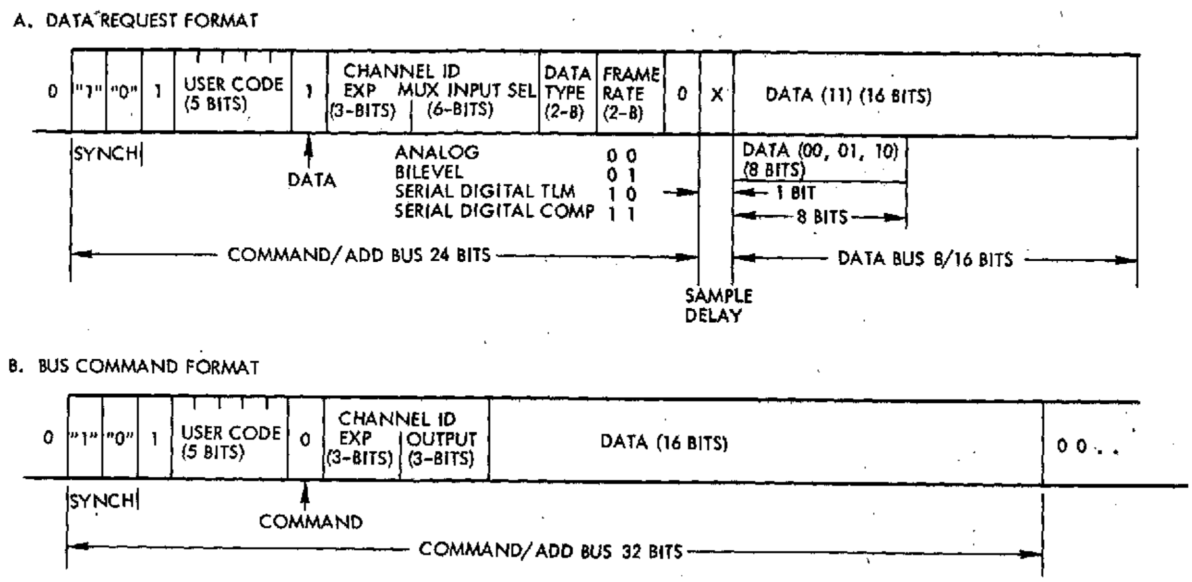
Figure 2-9. Data Handling Concept

The data bus is operated as a four-wire full duplex system. Computer commands are transmitted from the bus controller on one wire pair, and computer data and telemetry are received by the bus controller on the other wire pair. The bus bit rate is 1.024 Mbit/sec.

The format used for transmission of data on the bus is illustrated in Figure 2-10. As shown, 24 bits are used for addressing a data interface unit with a request for information; 32 bits are required for a command. Return data need no address because they are timed by the computer and bus controller. Word length is 8 bits for telemetry and 16 bits for computer data.

2.4.5 Safe Mode Bus

The safe mode bus enables the observatory to enter the safe mode (sun bathing) if a predefined failure occurs anywhere in the observatory. This is done by a line common to all modules that is normally held at a logical high level by a medium impedance source in the power module. An unsafe condition in any module can be used to ground this line (if provision was made prior to launch). All modules continually sense the level of this line and automatically enter their predetermined safe condition whenever the line goes low.



NOTES:
USER CODE (5 BITS) IDENTIFIES THE DIU OR MODULE
CHANNEL ID IDENTIFIES THE DATA REQUEST CHANNEL WITHIN THE DIU

Figure 2-10. Bus Formats

2.4.6 Spacecraft Harness

The spacecraft harness is dual redundant. Diode isolation at the load removes load faults and line breakers in the power module protecting against harness shorts. The breakers in the power module are magnetically driven and may be opened or reset by control signals. Power distribution to the spacecraft and payload is similar, except that the two breakers for the spacecraft dual redundant harness may not be opened simultaneously by signals through the data interface unit. Spacecraft power is completely disconnected only by hardwire control through the umbilical. This prevents inadvertant shutdown of the spacecraft in flight.

Individual spacecraft module harnesses fan out from the junction box near the power module. Harnesses to the modules in the payload fan out from a junction box near the transition ring in the payload structure. This permits separate assembly and test of spacecraft and payload systems. All modules obtain power through dedicated buses from the spacecraft junction box.

A separate heater bus is also routed to each module to maintain module temperatures during maintenance and resupply. This bus is routed to the umbilical at the spacecraft junction box so that removal of the power module cannot interrupt the minimal module temperature control required during resupply.

An input filter in each module controls harness noise signals and minimizes the amplitude of cold-start transients. The battery and main regulator filter have low dynamic impedance, and the common main power bus has minimum length; there is not significant dynamic coupling between modules.

2.4.7 Electromagnetic Compatibility

The replaceable spacecraft and payload modules must be mutually compatible during orbital operation and must be electromagnetically compatible with existing and future launch and resupply vehicles. This compatibility is achieved by limiting the magnitude of all extraneous electromagnetic emissions, by protecting all sensitive input signals and/or control circuitry from external influences, and by minimizing the number of input/output interfaces at both the observatory and module level.

Electrical system grounding concepts, case and cable shielding criteria, electrical bonding methods, and specification controls ensure compatibility and provide adequate margins between output interference emission and input susceptibility levels. Each module is a functionally independent operational unit with input/output filtering, shielding, or bandwidth limitations applied at its interfaces as needed. Thus, redesign of any mission-peculiar module will not have serious impact on the compatibility margins of other modules.

The primary DC power and party line data bus distribution subsystems have single-point grounds. Also, single-point grounding is used for low-level analog sensor circuits and high-current control circuits. With few exceptions, subsystem circuitry wholly contained within a module employs multiple-point grounding, using the module radiator panel and support structure as the ground reference plane and return path for secondary power and signal currents within the module.

2.4.8 Contamination Control

The observatory has several components particularly susceptible to contamination, either particulate or condensed volatiles: tracker optics, payload sensor optics, payload radiative cooler surfaces, and ultra precision bearings. In the case of spectro-radiometric instruments, degradation effects are particularly complex because deposits on optical surfaces are generally spectrally variant. This causes wavelength-dependent attenuation which, besides reducing overall sensitivity, affects the total optical train and is therefore difficult to calibrate.

After launch, internal calibration sources are limited. Deposits on optics that may be spatially distributed require an extended source at infinity, which is impractical with internal sources. Also, internal calibration lamps and associated optics have questionable spectral and intensity stability.

The sun has been used for calibration in the past, but it has orders of magnitude greater brightness than intended targets and has produced limited success. The moon has a stable albedo and approximately the proper brightness and may be preferred as a calibration source. The observatory could be commanded to view the full moon but at infrequent intervals (an opportunity occurs once every 28 days).

The Space Shuttle introduces new problems in contamination control that cannot be solved by conventional procedures, such as clean room conditions during spacecraft-to-launch-vehicle mating and jettisonable covers. Maintaining the Shuttle bay at a high level of cleanliness is impractical because of the short turnaround between missions and operational modes. Its reaction control thrusters and its many lubricated mechanisms are new problem areas. One suggestion is to include as part of the observatory flight support system an internal shroud or cocoon-like tent to protect the observatory. However, this imposes penalties in weight, cost, and complexity. We recommend commandable covers over all critical elements that can be closed when Shuttle is near the observatory. This adds complexity to the observatory but may be mandatory.

One special problem with covering optics is the need to vent the associated cavities during launch and retrieval. This can be done with purged gas or integral filters. Protection during retrieval is desirable since the accurate analysis of deposits accumulated on optics during extended space operation is an important data point in determining future contamination requirements.

2.5 REDUNDANCY AND RELIABILITY

The design concept of the modular spacecraft allows for a range of redundancy levels within each module, from nonredundant to multiple redundancy. Each module is configured to permit mission-selectable redundancy. The choice of redundancy for any particular mission depends on several factors:

- Mission length (e.g., 1 or 10 years)
- Mission maintenance approach (expendable or serviced)
- Payload design life (and obsolescence factors)
- Other design life factors
- Mission type (operational or R&D)

For a Shuttle-serviced observatory, servicing costs must be traded against spacecraft hardware costs. Four redundancy configurations have been defined for the spacecraft, ranging from the minimum level compatible with servicing to a more nominal level of redundancy, with most elements of the spacecraft backed up by a standby unit. Table 2-2 characterizes these redundancy configurations.

The basic (i.e., least expensive, lightest) configuration is the one with minimum redundancy for circular orbit servicing. Only the redundancy necessary to ensure against loss (i.e., unserviceability) of the spacecraft via a single failure is provided. This low degree of replication is made feasible by an attitude control safe mode, which sunpoints the spacecraft using hardware which is, for the most part, not employed during normal operations.

Increases in redundancy from the minimum level increase spacecraft weight and cost, but also increase mission duration prior to failure. Table 2-3 shows normalized cost and incremental weight as a function of mean mission duration (MMD) for several values of design life, where the data are based on the four point designs of Table 2-2 (this MMD does not consider payload failures). The baseline redundancy configuration presented in this report is that of Variant 2, which gives an MMTF of about 2 years.

Table 2-2. Redundancy Configurations

Configuration	Description	Redundant Equipment
Minimum	Redundancy necessary to assure no single-point failure preventing retrieval	• Power converters in attitude determination, electric power, and solar array and drive modules • Redundant array drive motor, electronics • Internally redundant power control/ regulation in electric power module
Variant 1	Least reliable elements made redundant; MTTF raised to about 18 months	• Redundancy of Minimum case • Redundant transfer assembly, star tracker, gyro in attitude determination module • Redundant on-board computer memory module • Redundant on-board computer central processor unit
Variant 2	Least reliable elements made redundant; MTTF raised to about 24 months	• Redundancy of Variant 1 • Redundant DIU, SCU in actuation module, attitude determination module, communication and data handling module, and electric power module • Redundant power converter in communication and data handling module • Redundant bus controller in communication and data handling module
Nominal	Most electronic assemblies made standby redundant; "typical" redundancy level for long-life satellites	• Redundancy of Variant 2 • Dodecahedron gyros (three in standby) • Redundant magnetometer • Redundant safe mode electronics • Redundant command receiver, telemetry transmitter, baseband assembly • Redundant DIU, SCU in solar array and drive module

Table 2-3. Properties of Redundancy Configurations

Redundancy Configuration	Mean Mission Duration (months)*					Normalized Cost	Incremental Weight (lb)
	$T_D = 1$ yr	$T_D = 2$ yr	$T_D = 3$ yr	$T_D = 4$ yr	$T_D = 5$ yr		
Minimum	6.7	8.5	8.9	9.1	9.1	1.00	0
Variant 1	9.5	14.7	17.2	18.3	18.8	1.07	55.8
Variant 2	10.5	17.7	22.0	24.3	25.5	1.11	75.8
Nominal	11.5	21.5	29.9	36.2	40.8	1.18	112.1

*T_D = Design life.

2.6 COMMUNICATIONS AND DATA HANDLING MODULE DESIGN

The communications and data handling (CDH) module (see Figure 2-11) interfaces with the NASA ground stations and serves as the information center for all on-board communications. It contains the equipment required to receive, demodulate, and process uplink command information; collect, process, and telemeter housekeeping and medium-rate user data; coherently transpond range information; and centrally perform on-board computations. It also contains equipment that implements and controls the spacecraft data bus.

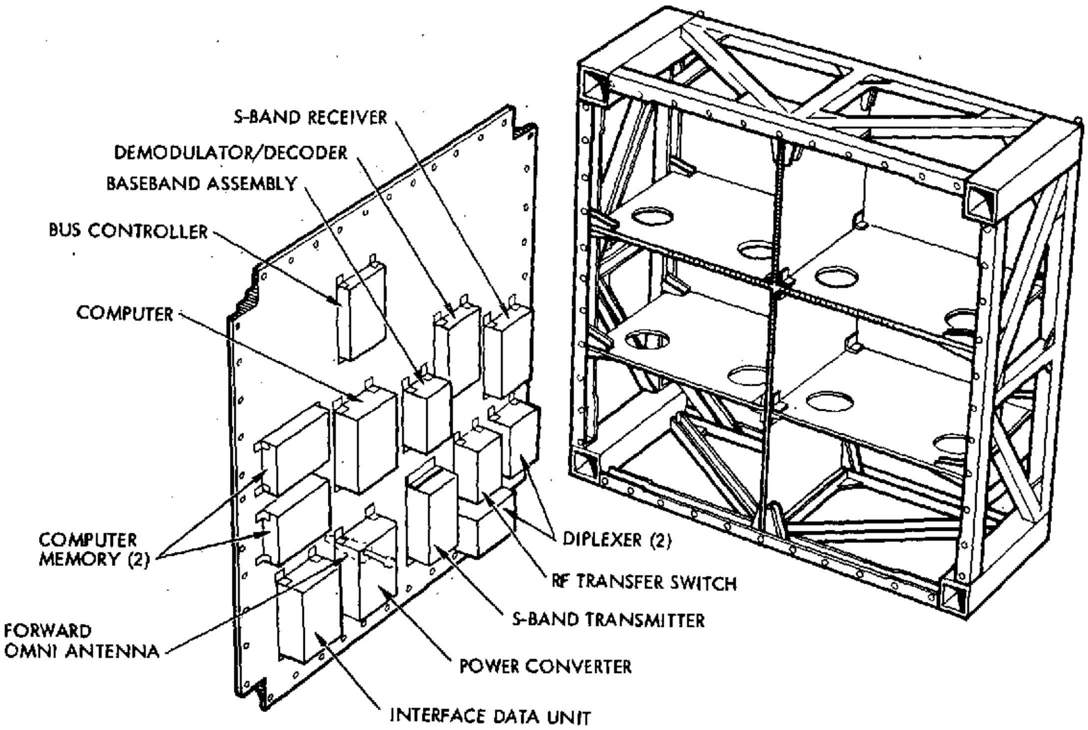


Figure 2-11. Communications and Data Handling Module

Table 2-4 lists the capabilities of the baseline CDH module design; Figure 2-12 is a block diagram.

The CDH design uses proven spacecraft equipment and technology to achieve modularity. It has adequate volume and flexibility to accommodate increased redundancy, magnetic tape recorders, expanded computer memory, and other requirements of future missions. TDRSS capability is achieved by adding special mission equipment to the payload complement with no modification to the CDH module. The CDH can accommodate

Table 2-4. CDH Capabilities

Capability

RF Group

Communications fully compatible with "GSFC Aerospace Data Systems Standards", X-560-63-2

Antenna Polarization

Dual right and left hand circular polarized omni-antennas provided

Transmit Frequency

Carrier lies in 2200 to 2300 MHz range

Receive Frequency

Carrier lies in 2050 to 2150 MHz range

RF Characteristics

- Transmit carrier stability is one part in 10^5
- Transponder ratio: 221/240
- Transponder sidetone frequency: 500 kHz
- Command bit rate: 2000 bits/sec
- Command modulation: PCM, split-phase-M command data frequency modulates 70 kHz subcarrier. Subcarrier phase modulates carrier.
- Narrowband telemetry rate: Selectable: 32, 16, 8, 4, 2, 1 kbit/sec
- Narrowband telemetry modulation: PCM-split-phase/PSK/PM on 1.024 MHz subcarrier
- Medium rate telemetry: 500 kbit/sec maximum
- Medium rate telemetry modulation: Split-phase PCM/PM (direct carrier modulation)
- Transmitter power: 2 watts
- Telemetry data coding: Manchester (split-phase)

Capability

Receiver/Demodulator Options

- Baseline Design: Single receiver/demodulator with RHCP and LHCP antenna diversity combining
- Alternate Design: Dual receiver/demodulators with RHCP and LHCP antenna diversity combining

Command System Performance

- Probability of false command execution less than 1×10^{-19} for input levels of -112 dBm and above
- Probability of good command rejection less than 1×10^{-5} for input levels of -112 dBm and above

Receiver Combining

Diversity combining plus receiver squelch ensures highest output signal level into demodulator/decoder for dual receiver operation

Transmitter

S-band transmitter capable of simultaneously transmitting 32 kbit/sec of real-time housekeeping data and 512 kbit/sec of medium rate data

Transmitter Data Selection

Four selectable medium rate modes:

1) Ranging tones (500 kHz)
2) Tape recorder dump (512 kbit/sec)
3) Special instrument dump (512 kbit/sec)
4) Computer memory dump is time division multiplexed into 32 kbit/sec housekeeping data

RF Group-Receiver/Demodulator

Active redundancy provided for dual receivers/demodulators

RF Switch

RF switch provided to switch transmitter to either antenna

Spherical Antenna Coverage

97 percent spherical antenna coverage provided at -1.0 dBi

Table 2-4. CDH Capabilities (Continued)

Capability

Worst Case Command Link Margins:

Carrier: 49.2 dB
Command: 47.3 dB (10^{-6} BER)

Worst Case Telemetry Link Margins:

Mode 1: 32 kbit/sec housekeeping plus 512 kbit/sec medium rate data:

- Carrier: 33.8 dB
- 32 kbit/sec: 7.3 dB (10^{-6} BER)
- 512 kbit/sec: 8.4 dB (10^{-6} BER)

Mode 2: 32 kbit/sec housekeeping plus 500 kHz tone ranging:

- Carrier: 38.5 dB
- 32 kbit/sec: 20.5 dB (10^{-6} BER)
- Ranging: 18.3 dB (5 meter, 1σ)

Input Command Message

Message length: 43 bits
Bit rate: 2 kbit/sec
Format:
- Introduction - 200 bits
- Phase - 1 bit
- Spacecraft address - 7 bits
- Op. code - 2 bits
- User code: 5 bits
- Channel ID - 6 bits
- Data - 16 bits
- Check code: 7 bits

Bus Configuration

Full duplex, two separate buses 1.024 Mbit/sec, flexible slot assignments

Command/address bus
- Command format: 32 bits
- Data Request format: 24 bits

Data Bus
- Telemetry data: 8-bit words
- Computer data:
  - Analog bilevel: 8-bit words
  - Serial digital: 16-bit words

Data clock derived from continuous Manchester code

Capability

Remote Units

- Maximum number of units: 32
- Input power: +28 volts from primary power bus
- Remote multiplexer: 64 data channel/data interface unit-expandable to 512 data channels by addition of up to seven expanders
- Remote decoder

  Command outputs:
  32 pulse
  7 serial digital
  can be expanded to 256 pulse commands/data interface unit, 56 serial digital by addition of up to seven expanders

On-board Computer

- Type: general-purpose, digital binary, full parallel organization, one index register
- Arithmetic:
  - Fixed point binary
  - 16-bit data word (min)
  - Add instruction <5μsec
  - Multiply instruction
  - Divide instruction
  - Double precision add <40 sec
  - Double precision multiply <200 sec
  - Four logic instructions: and, or, exclusive or, complement
  - conditional/unconditional transfers
- Memory
  - Nonvolatile, expandable to 64K words in 8K word modules
  - Cycle by cycle power switching of 8K memory modules (100 MW maximum increase in standby power per 8K module)
  - Protection against illegal write to memory
- Input/Output
  - 16 maskable interrupts
  - All input/output channels must be able to operate in both DMA mode and program control mode

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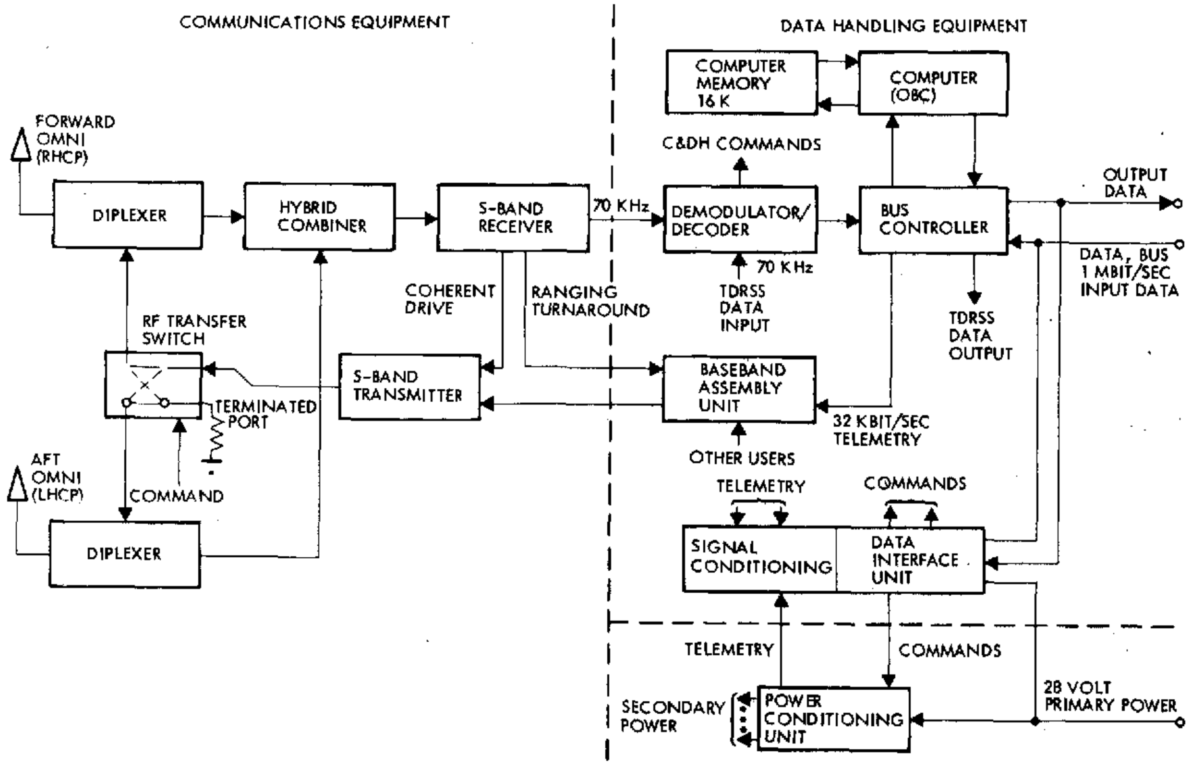


Figure 2-12. Communications and Data Handling Module Baseline Data

medium-rate data users having rates to 512 kbit/sec with no impact. If tape recorders are used, they can be dumped at this same rate. In the baseline configuration, housekeeping telemetry data are compacted and stored in the on-board computer, obviating a housekeeping data tape recorder.

The communication equipment consists of a diplexer, receiver transmitter/modulator, RF switch, baseband assembly, aft and forward omni antennas. The data handling equipment consists of demodulator/decoder, bus controller, data interface units, on-board computer, and the central clock. (The central clock is supplied by the bus controller crystal oscillator.)

2.6.1 Communications Equipment

Command reception and telemetry transmission are achieved via two omni-directional antennas, which provide over 97 percent spherical coverage measured relative to the -1.0 dBi point. The output from each antenna is coupled to a hybrid combining network by two diplexers. The hybrid combines the two signals from the diplexers to provide a signal output to an S-band receiver tuned to a frequency within the 2050 to 2150 MHz range. One output of the hybrid network is terminated. One of the antennas is right-hand circularly polarized and is on the earth-pointing side of the spacecraft; the other is left-hand circularly polarized and is on the opposite or aft side. Thus, the STDN can command the spacecraft via one of the two antennas, regardless of how the spacecraft is oriented.

The input to the S-band receiver contains command and ranging information in unified S-band (USB), STDN-compatible form. The receiver detects this information and provides the command data as an output to a demodulator/decoder unit, which is in the data handling portion of the module. The demodulator/decoder input from the receiver consists of a 70 kHz subcarrier containing 2 kbit/sec command data. The detected ranging data is output from the receiver to a baseband assembly unit, also part of the data handling equipment. This unit sums the range data with a 32 kbit/sec, biphase-modulated, 1.024 MHz telemetry subcarrier and a 512 kbit/sec (medium-rate date) direct digital data stream to form the baseband signal which phase modulates the downlink RF carrier.

A frequency reference, coherent with the uplink carrier, is output from the receiver to the S-band transmitter. This reference, suitably multiplied in frequency, is phase modulated by the composite baseband signal from the baseband assembly unit. Because the downlink carrier is phase coherent with the uplink (with a frequency ratio of 240/221), two-way range rate or doppler information is provided. The S-band transmitter outputs a 2-watt RF signal, which is coupled to one of the two omni antennas by an RF transfer switch.

Advanced missions may require operation at higher orbit altitudes with higher data rates and higher system reliability. For missions requiring higher reliability, equipment can be added and cross-strapped. For example, a redundant receiver can be added by simply connecting it to the terminated port of the hybrid combiner in Figure 2-12. A redundant transmitter can be accommodated by connecting its output to the terminated port of the transfer switch. Missions that require a higher EIRP also can be accommodated by mounting a fixed, 2-foot parabolic antenna to the module and connecting it via an RF switch to the 2-watt transmitter.

The baseline design was established using STDN/USB compatible uplink and downlink carrier and subcarrier frequencies, data rates, and modulation formats. Maximum real-time telemetry data rate is 32 kbit/sec; maximum medium data rate is 512 kbit/sec; and a maximum uplink data rate is 2 kbit/sec. Since these rates are maximum for the spacecraft, the CDH can accommodate other low altitude requirements (300 to 900 nautical miles). In general, performance is more than adequate for missions having an orbit altitude of 1810 kilometers (975 nautical miles) or less. A synchronous mission (such as SEOS) requires an earth coverage 2-foot antenna, fixed-mounted on the module for operation with the baseline 2-watt transmitter. Table 2-5 lists design modifications required for selected missions.

Table 2-5. Post-EOS-A Missions and Design Impacts

Mission	Orbit Altitude (km)	Uplink Antenna	Downlink Antenna	Downlink Transmitter	Weight Impact	Power Impact
SEOS	36,041 (synchronous)	Omni	2-foot dish	2 watts	2.5 lbs	None
SEASAT-A	725	Omni	Omni	↑	None	None
Solar Maximum Mission (SMM)	556	Omni	Omni	│	None	None
Gap Filler (5-Band MSS)	861	Omni	Omni	│	None	None
SEASAT-B	723	Omni	Omni	↓	None	None
Advanced SMM	556	Omni	Omni	2 watts	None	None

2.6.2 Data Handling Equipment

The data handling equipment complement is shown in Figure 2-13. The demodulator/decoder accepts a 70-kHz subcarrier from the S-band receiver, demodulates it, and transmits a serial data stream to the bus controller. The bus controller supervises communication on the data bus; i.e., it initiates command and interrogation signals to the various digital interface units (DIU's) contained in other modules and accepts telemetry and computer data returned in response. The DIU's contain a remote command decoder, which transfers serial or pulse commands to the user equipment. The DIU also contains a PCM encoder for time-division multiplexing both analog and digital data.

The bus controller is the CDH module interface with the computer, accessing the memory directly through the DMA. Two kinds of data are transferred to the computer memory: data from the demodulator/decoder tagged for storage in the computer memory and data requested by the computer from the data bus. The bus controller also provides backup circuits for the telemetry formatting normally done in the computer.

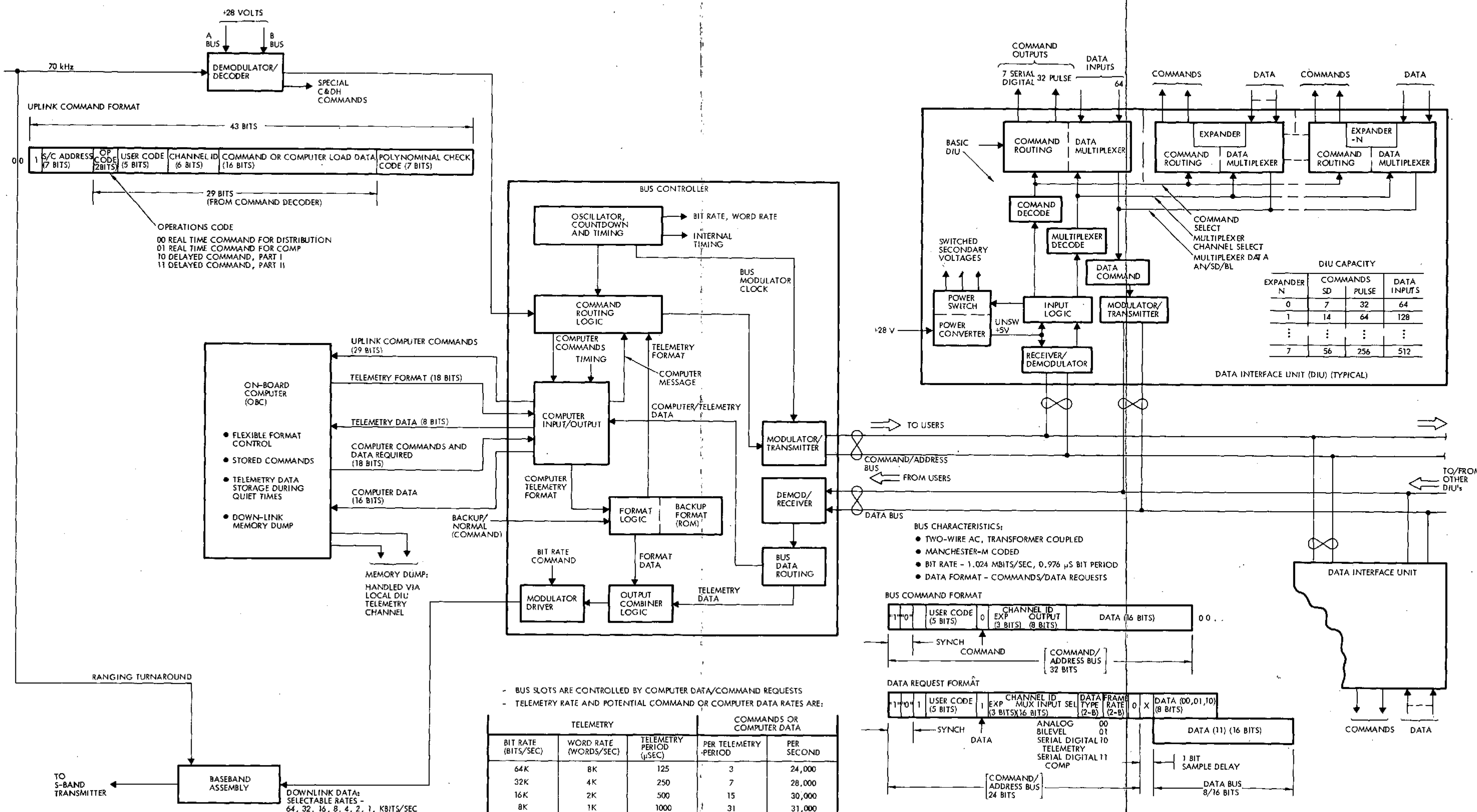
The computer performs many functions (e.g., telemetry formatting, command and telemetry storage, thermal control, and attitude control system calculations). The baseline computer is nonredundant and operates with a 16,000 word memory.

The baseband assembly provides the subcarrier for the downlink telemetry and sums all downlink data for transmitter modulation.

The data handling equipment can handle many advanced missions without design change:

- Demodulator Decoder. The principal requirement is the command data rate. The command word is 44 bits long: 15 of these are used for overhead, spacecraft address, priority check. A 29-bit word is transmitted to the bus control unit. The uplink bit word is transmitted to the bus control unit. The uplink bit rate is 2 kbit/sec, including overhead; the net bit rate (maximum) is 1.3 kbit/sec. No future mission is expected to exceed this requirement.

- Bus Controller. The bus controller operates with the data bus at a bit rate of 1 Mbit/sec. No future mission should require full occupancy of the data bus.



- BUS SLOTS ARE CONTROLLED BY COMPUTER DATA/COMMAND REQUESTS
- TELEMETRY RATE AND POTENTIAL COMMAND OR COMPUTER DATA RATES ARE:

TELEMETRY			COMMANDS OR COMPUTER DATA	
BIT RATE (BITS/SEC)	WORD RATE (WORDS/SEC)	TELEMETRY PERIOD (μSEC)	PER TELEMETRY PERIOD	PER SECOND
64K	8K	125	3	24,000
32K	4K	250	7	28,000
16K	2K	500	15	30,000
8K	1K	1000	31	31,000
4K	500	2000	63	31,500
2K	250	4000	127	31,750
1K	125	8000	255	31,875

Figure 2-13. Data Handling Equipment Block Diagram

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- Computer and Memory. The present 16K memory can be expanded to as high as 64K. The computer should be adequate for all future missions and for missions requiring redundancy.

- DIU. Single slice capability is 7 serial commands, 32 pulse commands, and 64 telemetry data inputs. Eight expanders, each containing the same number of functions as the initial slice, may be added to any unit. As many as 32 DIU's can be handled by one bus controller. No future spacecraft or module should exceed this expandable capability.

- Baseband Assembly. The baseband assembly has four spare summation amplifier inputs as well as associated switching capability. No advanced mission should exceed this capability.

2.7 ATTITUDE DETERMINATION MODULE DESIGN

A key feature of the attitude control design for the standard spacecraft bus is grouping attitude sensing functions within an attitude determination module, which is mission-independent, and grouping actuation functions, including propulsion, into an actuation module, which can change from mission to mission if required. This section describes the attitude determination module design (Figure 2-14). Section 2.8 covers the actuation module. Both modules are applicable to a range of missions. Software functions are mechanized in the on-board computer within the communications and data handling module.

The attitude determination module (ADM) (see Figures 2-15 and 2-16) can be used without change for a wide variety of orbits for both earth and inertial pointing. It employs existing state-of-the-art components, including a three-axis rate integrating gyro package and two body-fixed star trackers. These sensors provide periodic gyro rate-bias updates. Sensor information is processed by the on-board computer, which integrates the gyro rates to provide attitude and establishes optimal attitude estimates at the star tracker update times.

Figure 2-17 illustrates the star tracker geometry in an earth-pointing mission. Normal operation requires two star trackers. A third star tracker is optional and is used as a standby unit. The star trackers are positioned on the spacecraft so that a large swath of many stars is viewed on the celestial sphere with minimum radiant interference from the sun. Each tracker scans the same swath minimizing the on-board star catalog requirements.

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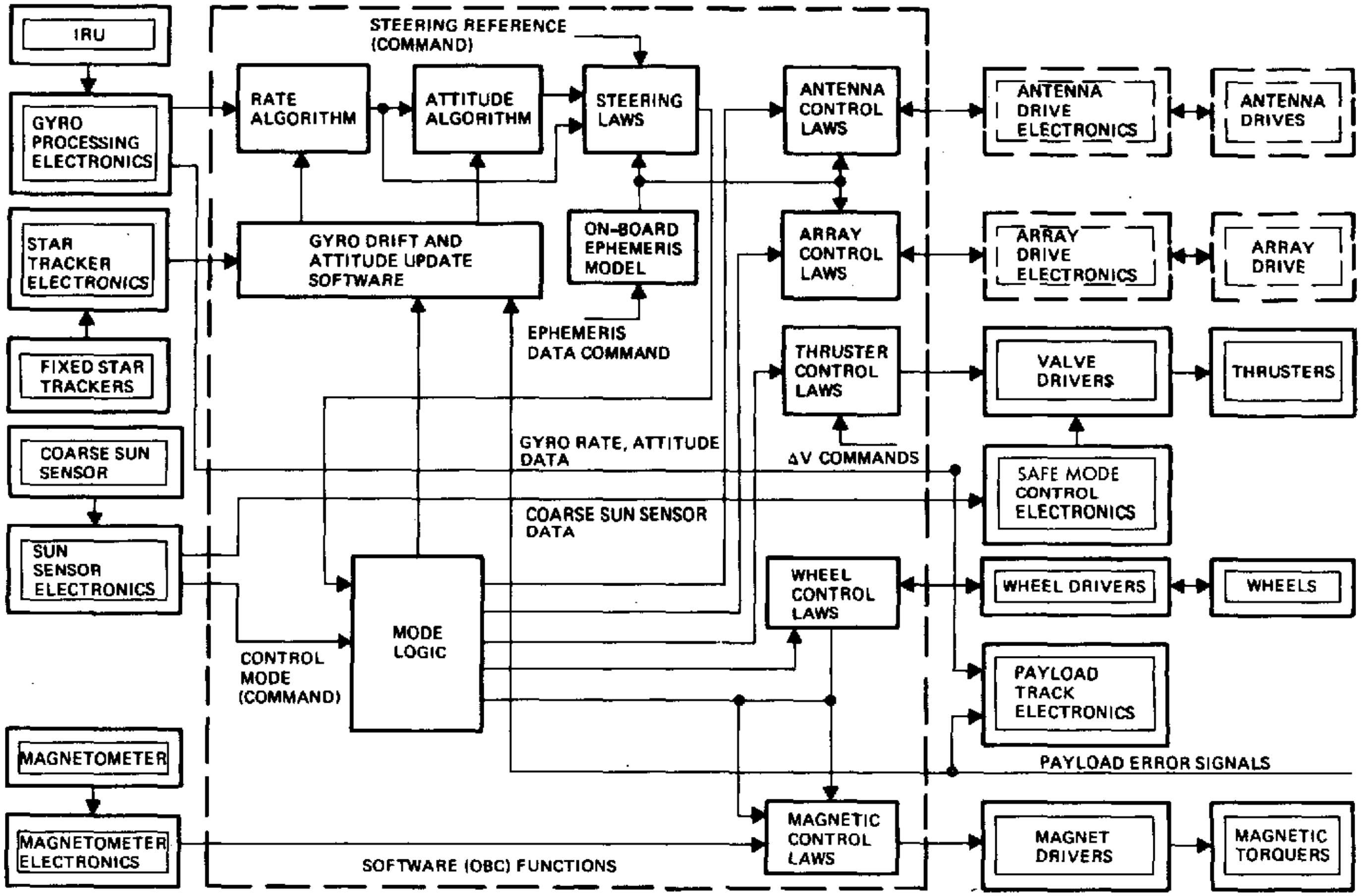


Figure 2-14. Attitude Control Functional Block Diagram

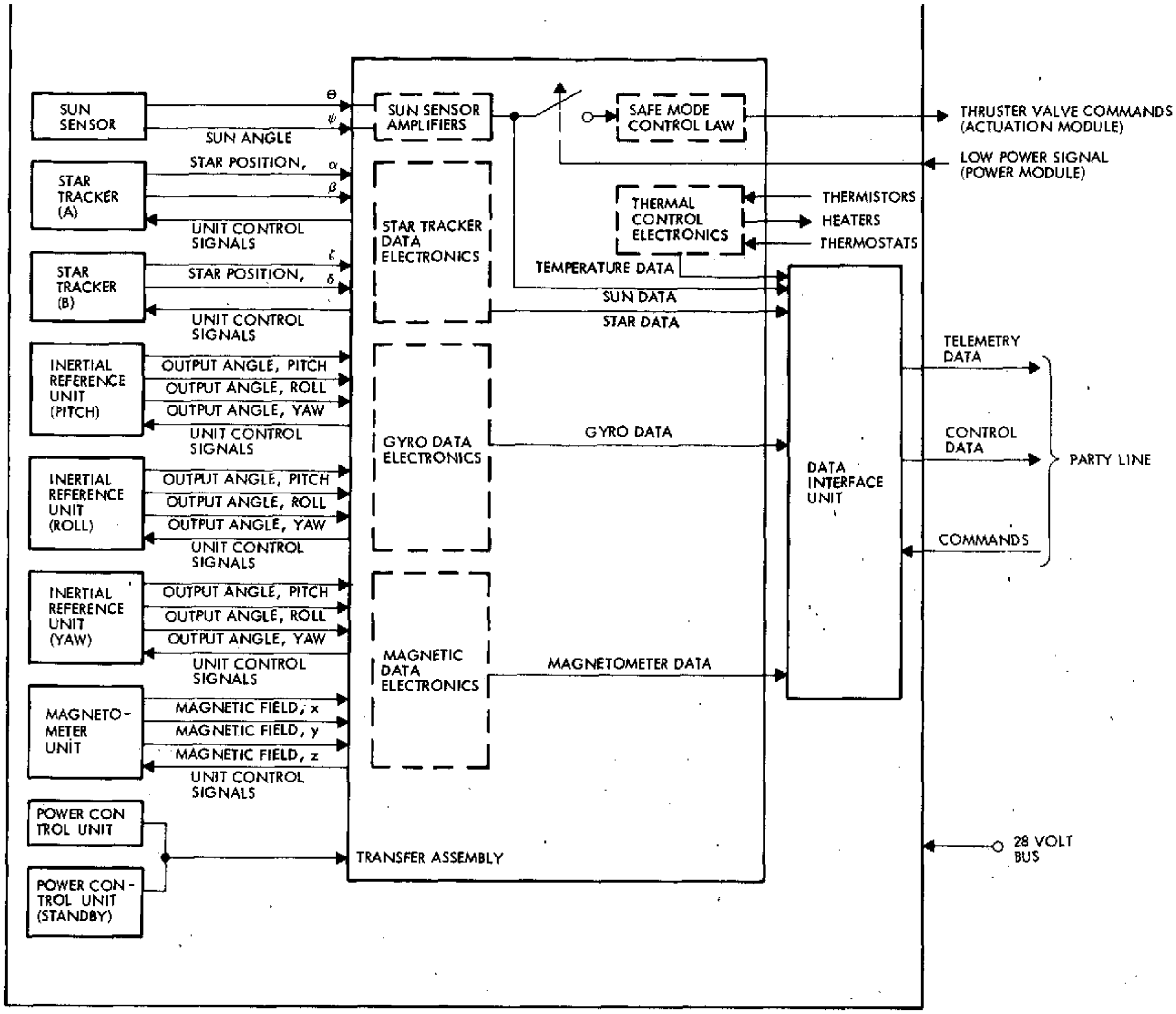


Figure 2-15. Attitude Determination Module Block Diagram

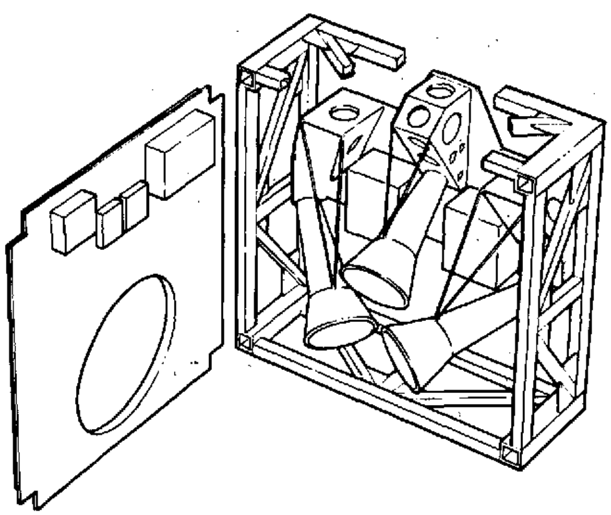
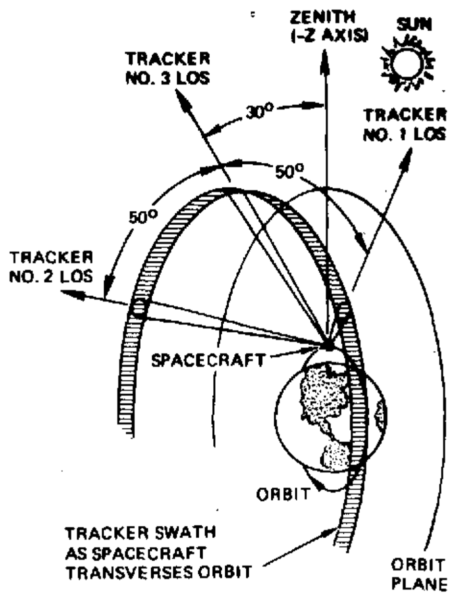


Figure 2-16

Attitude Determination Module



- STAR TRACKER FIELD OF VIEW: 8 x 8 DEG
- STAR MAGNITUDE: +2 TO +6

Figure 2-17

Star Tracker Geometry

Attitude estimation during earth-pointing is performed by an on-board computer, which receives IRU data input every 200 msec and star tracker information every 3 to 10 minutes. The IRU-measured gyro rates are corrected within the computer for their bias drift rates, and the result kinematically integrated to yield an inertial attitude estimate. When a predetermined star enters a tracker field of view, a symbolic switch closes, and both the inertial attitude and the gyro drift rates are updated. Ground ephemeris data are provided periodically to a time-dependent ephemeris model which, along with the inertial attitude estimate, yields an estimate of spacecraft attitude in geocentric coordinates. These data are issued in the earth-pointing mode to steer the spacecraft and in the pitch mode to maintain the pitch axis normal to the apparent ground track. Yaw steering maintains the payload scan plane normal to the image velocity vector.

2.8 ACTUATION MODULE DESIGN

The actuation module consists of wheels, torquers, a hydrazine orbit transfer/adjust propulsion system, and a cold gas nitrogen reaction-control propulsion system.

Module design is shown in Figure 2-18. Figure 2-19 is a functional block diagram. Characteristics of the reaction wheels and magnetic torquers for the baseline design are as follows:

Reaction Wheels (3):

- Momentum: 7.2 ft-lb-sec at 1250 rpm
- Torque: 20 in.-oz
- Power: 53 watts (peak)

Magnetic Torquers (3):

- Magnetic field strength (linear): 120,000 pole-cm
- Power at 120,000 pole-cm: 0.5 watt
- Length: 40 in.

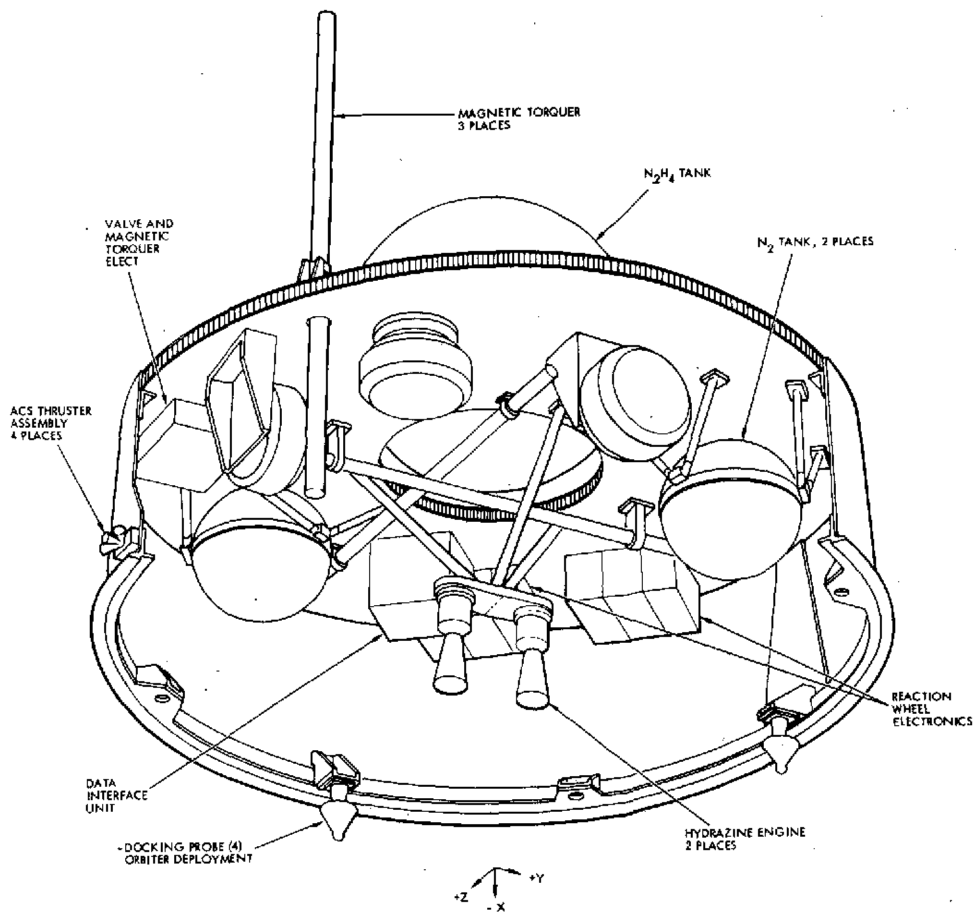


Figure 2-18. Actuation Module (piping not shown)

- Weight: 8.7 lb
- Coil: 40,300 turns (No. 32 copper wire)
- Residual magnetism: 688 pole-cm
- Core material: Allegheny Ludlum electrical steel 4750

For other missions, wheels as large as 150 ft-lb-sec storage capacity and 80 in.-oz torque can be accommodated. Similarly, magnetic torques to 10^6 pole-cm can be used to compensate for up to 0.01 ft-lb secular disturbance torque.

Propulsion functions of the module include orbit transfer, orbit adjust, and attitude control. Propulsion hardware is integral to the actuation module. Because the module is structurally and thermally self-contained, components and propellant may be added or deleted (within limits of weight and envelope) with little effect on overall spacecraft design. Propellant mass can be increased slightly by increasing the blowdown ratio for the hydrazine or by increasing the gas pressure for the cold gas. The modular design also allows changes in tank size and number of tanks to accommodate large changes in propellant loads.

Propulsion hardware includes a monopropellant hydrazine subsystem for orbit transfer and adjust functions and a dual level cold gas (GN_2) subsystem for reaction control. The hydrazine element can be deleted when velocity control is not required. The two elements are combined so that certain components can be shared to save cost and weight.

Missions not requiring orbit transfer can be achieved with a smaller hydrazine tank and lower thrust hydrazine thrusters. The reduced disturbance torques then allow single level reaction control.

In general, the storage capacity of the module is adequate for worst-case mission requirements. For missions where the requirement is lower, the excess capacity can be used to increase mission life and flexibility, or off-loaded to increase payload capability.

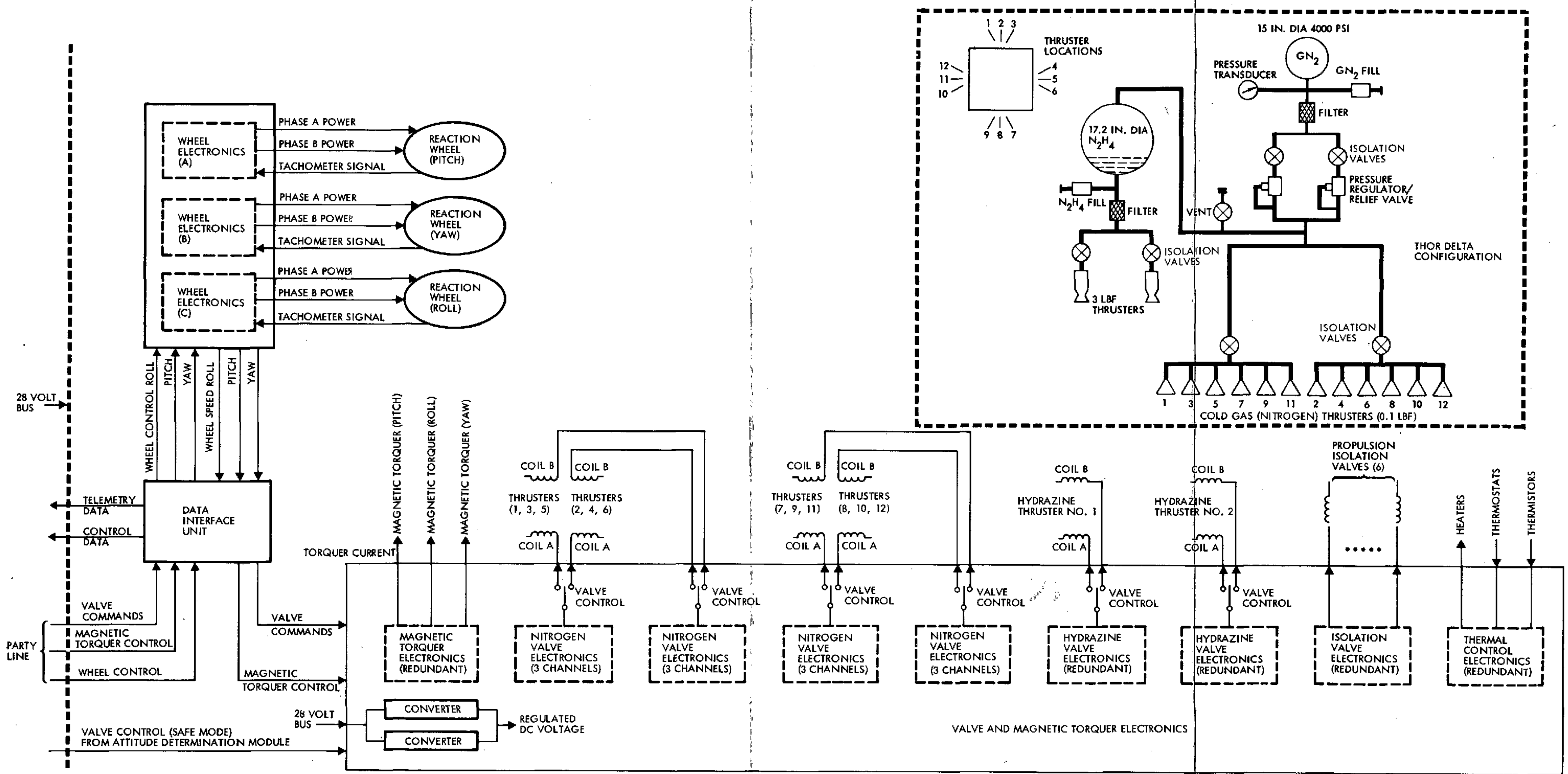


Figure 2-19. Actuation Module (Minimum Redundancy)

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2.9 POWER MODULE

Orbit altitude and inclination have a significant effect on the size of electrical power components, particularly solar arrays and batteries. For example, the power required for battery recharge at low altitudes is approximately 33 percent of the end-of-mission solar array power; it is only 15 percent at geostationary altitude. Solar array maximum temperatures are approximately 15 degrees greater at low altitudes than at geostationary altitudes because of thermal heat inputs from the earth.

To accommodate the range of load power levels and orbit profiles with minimum reconfiguration and adjustment, the electrical power configuration of the standard spacecraft bus is subdivided into two modules: power and solar array and drive. Although both modules are mission-dependent, they contain standard battery and solar array submodules that can be added or deleted in increments to suit the mission with little or no impact on other modules or an overall spacecraft design. The remainder of this section deals with the power module; Section 2-10 deals with the solar array module.

The power module performs primary power regulation, fault protection, and energy storage. Raw power is conducted to the power module from the solar array and drive module through four twisted pairs. Regulated power flows through the power module to the spacecraft and payload modules via the spacecraft function box and harness. The major elements of the power module are the power control unit, battery assembly, secondary power and bus protection assembly, diode assembly, power disconnect assembly and data interface unit (see Figures 2-20 and 2-21).

The power module contains minimum of two (baseline configuration) and a maximum of four 40 amp-hr batteries; one battery cell size is used for all missions. Table 2-6 summarizes battery characteristics for the baseline spacecraft.

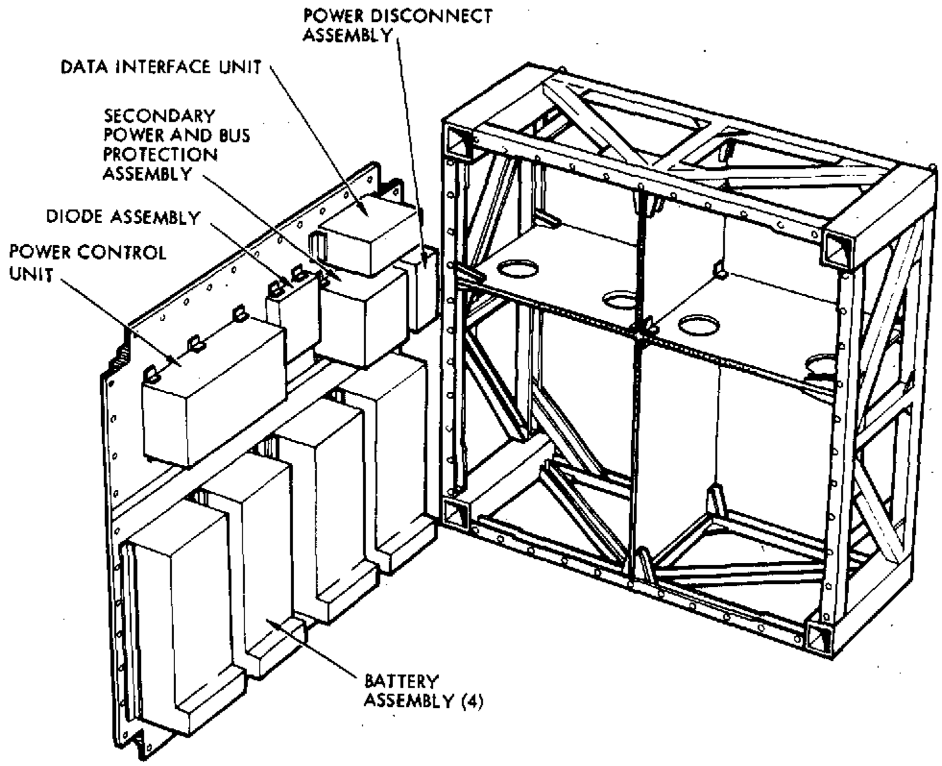

Figure 2-20. The Power Module Can Accommodate Up to Four Batteries; The Baseline Spacecraft Uses a 2-Battery Configuration

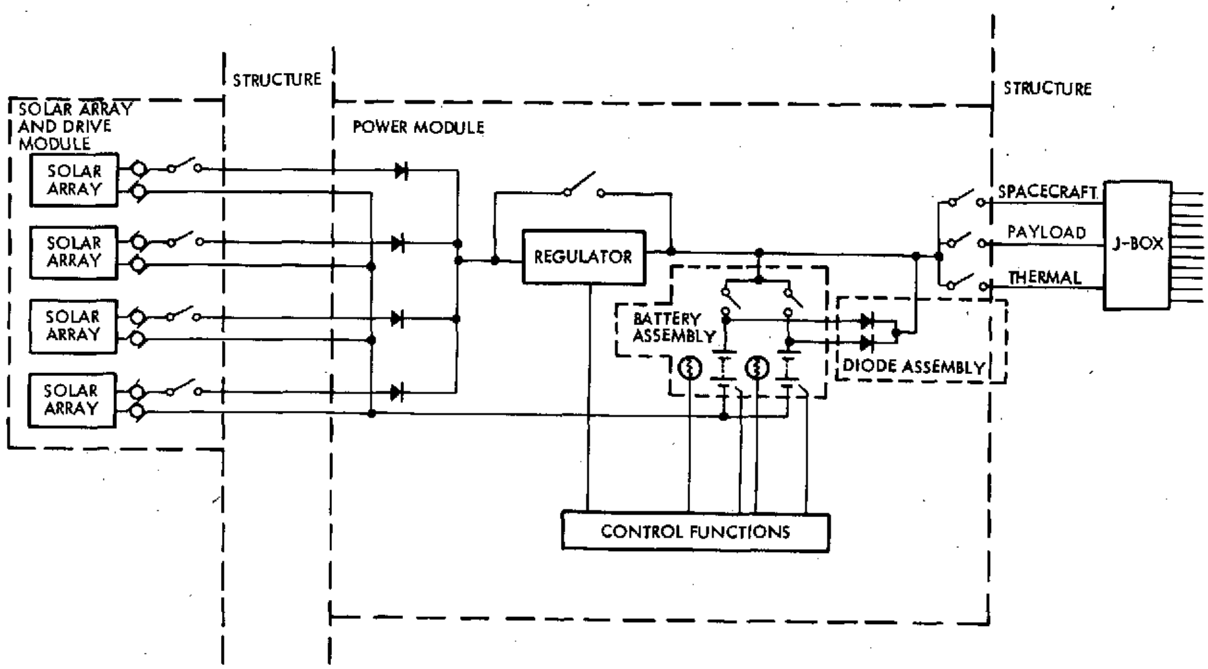

Figure 2-21. Power Module and Interfaces

Table 2-6. Battery Characteristics for Baseline Spacecraft

Total battery system capacity	80 amp-hr
Battery cell capacity	40 amp-hr
Batteries per spacecraft	2
Cells per battery	22
Maximum depth of discharge	15.2%
Total delivered energy	333.3 watt-hr
Weight per battery	104.5 lb
Total battery weight	209.0 lb

The battery design must be compatible with missions for which load characteristics vary over a wide range. Planned future missions will include loads with high peak power and low duty cycle. This type of load has considerable impact on the design of battery discharge regulators. To avoid the cost and complexity of discharge regulators and to supply peak loads with minimum line transients, batteries are discharged through diodes or directly to the main bus.

The power module uses a parallel charging design that was flight-qualified on the Orbiting Astronomical Observatory. This charge approach can be adapted to batteries of widely differing size in orbits ranging from low to geostationary altitude. Efficiency is high because at end of life the power flows directly from the solar array to the battery without a charge control loss. In addition, there is less equipment to build and test than a system that uses a charger with each battery. The advantages of lower hardware costs and lower power losses more than offset the need for battery cell matching.

Batteries must be very uniform and similar in temperature and capacity. The temperature of each battery cannot vary by more than $\pm 2.5°F$ from the average temperature of all batteries. This requirement is greatly mitigated because the power module carries four batteries at the most and only two in the baseline. Batteries are mounted on a common high-conductivity plate to equalize temperatures during charging cycle. Each battery has one cell that detects overcharge and fault conditions. Each battery also contains three temperature sensing circuits for BVLS charge control and for the high temperature BLV switches (85 and 95°F).

The power control unit controls the battery state of charge throughout the three orbital charge/discharge routines: eclipse, initial sunlit period, and final sunlit period. During normal operations, battery enable/ disable switches and power disconnect switches are closed; during abnormal conditions the appropriate switch is opened. Main bus power is controlled by a pulsewidth modulation buck regulator. Fault sensing circuitry controls redundant regulator switching and provides safe mode enable signals.

2.10 SOLAR ARRAY AND DRIVE MODULE

The solar array and drive module performs the following:

- Power generation
- Solar array orientation
- Spacecraft structure heater control

The solar array and drive module generates all power during periods of illumination and maintains array orientation to sun-normal within ±10 degrees. The solar array can accommodate observatory loads between 0.3 and 2.0 kw by addition or removal of array subpanels, which are identical and interchangeable. This approach reduces cost because only one design need by qualified, only one set of test equipment and fixtures is needed, and the sections can be produced in unit lots for a wide range of missions, duty cycles, and orbits. Nominal end of life subpanel power is 40 watts.

The solar array and drive module comprises solar array, array drive and slip rings, drive electronics (redundant) power converter (redundant) for the solar array drive electronics, data interface unit, bus protection assembly, and power disconnect assembly (see Figures 2-22 and 2-23). The drive is a modification of the TRW COMSAT drive.

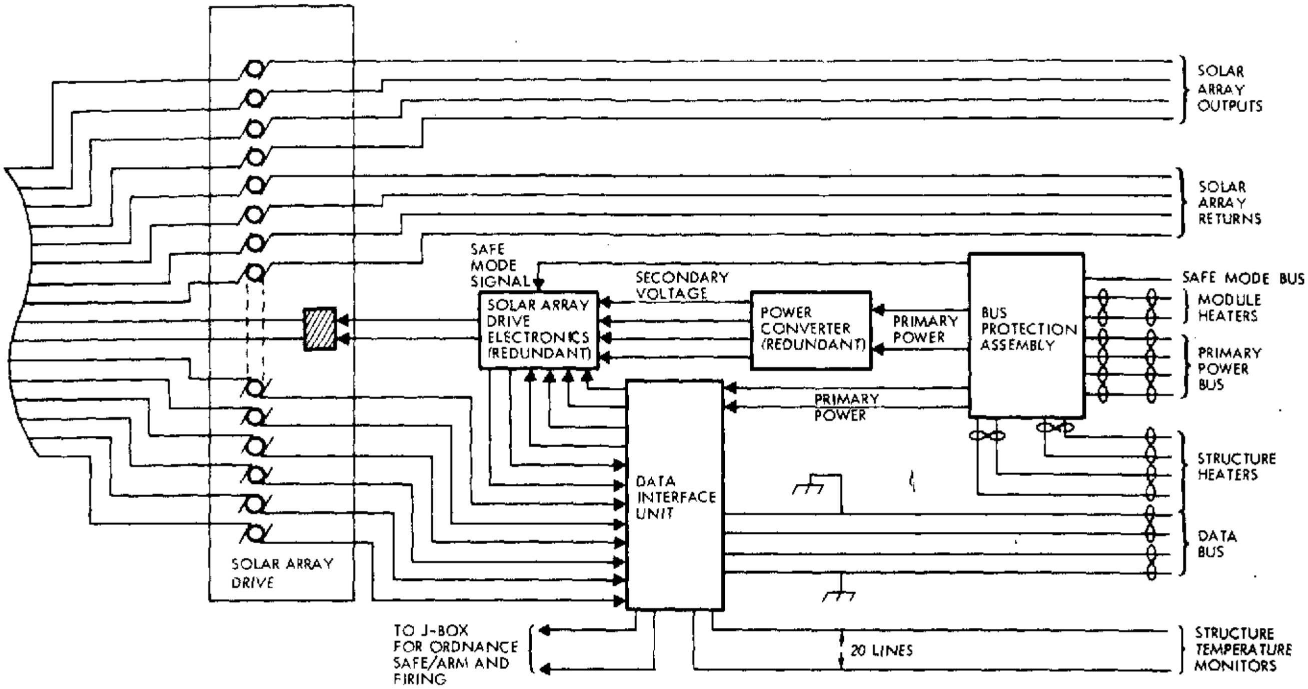

Figure 2-22. Solar Array and Drive Module Block Diagram

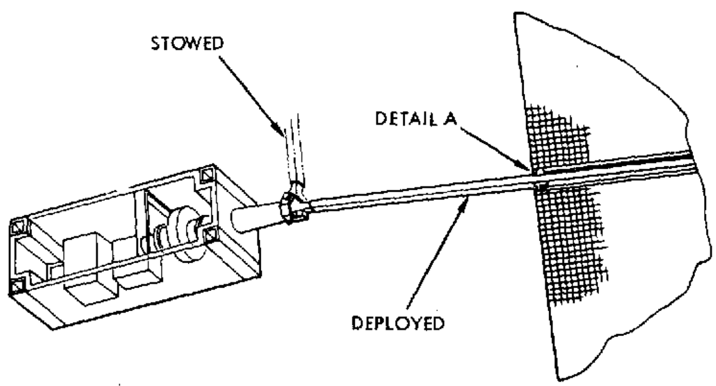

Figure 2-23. Solar Array and Drive Module

Solar array power is transmitted through the array-drive assembly and the power disconnect assembly to the power module. For the baseline design, the solar array is divided electrically into four panels which fold together for stowage during launch. In turn each panel consists of six subpannels. Each section delivers power to the drive interface by means of a power line and a return line. Thus, the array drive contains eight slip ring assemblies for power handling. In addition, it has seven slip ring assemblies for solar array temperature telemetry signals. Each slip ring assembly contains redundant brushes.

The power disconnect assembly disconnects the solar array from the electrical distribution system for spacecraft servicing in orbit by Shuttle. The power can be reopened while the observatory is docked to the Shuttle only on command transmitted through the Shuttle umbilical.

The basic building block used to adjust the solar array configuration for a range of missions is a honeycomb subpanel upon which solar cells are bonded. This design was flight-qualified for the five-year geostationary-altitude USN/USAF FLTSATCOM solar array.

2.11 SPACECRAFT STRUCTURE

The spacecraft is that part of the observatory from the transition ring aft, plus the solar array and drive module, which is just forward of the transition ring. The spacecraft comprises a primary structure, a group of modules, and mechanisms to release, separate, and deploy various spacecraft and payload devices.

The primary structure has six subassemblies:

- Equilateral triangular mainframe outlined by three longerons projecting aft from the plane of the transition ring
- Aft stiffened cylindrical shell structure which mounts the forward section of the separation joint
- An I-section transition ring and sandwich bulkhead in the plane of the ring, plus load fittings for Shuttle compatibility
- Six struts joining the transition ring to the forward end of the aft cylindrical shell structure
- Structure thermal control consisting of insulation, heaters and temperature sensors
- Structure harness assembly

The spacecraft structure (Figure 2-24) incorporates generous safety margins to minimize the need for detailed analytical loads prediction. With these margins, costly static and modal survey testing can be replaced with economical and effective three-axis sinusoidal base excitation tests. The primary structural material is aluminum. In local areas, titanium and/or fiberglass is used for thermal isolation because the thermal conductivity of these materials is significantly less than aluminum.

Three modules (Figure 2-25), the power, communications, and attitude determination, are similarly configured: each is 48 x 48 x 18 inches deep and consists of:

- Four square tubes at corners for mounting the latching mechanism
- Four identical numerically-controlled machined side frames (one each side and top and bottom)
- Equipment mounting/radiator sandwich panels on outboard surface
- Corner braces
- Internal bulkheads as required for individual equipment complements

The modules are cantilevered from the four corners at the inboard face of the module. Thus, the square tubes carry the primary load from the outboard equipment platform to the latch mechanism. The cross-braced frames reinforce the corner tubes by feeding the equipment panel loads into them via the diagonals. Once the frames and tubes are assembled, the module becomes a four-sided box with the sandwich equipment panel as a lid.

The corner braces reinforce the inboard plane stiffness of the module against shear deflection. The sandwich bulkheads, positioned for specific module equipment layout, provides intermediate support to the equipment mounting panel. Primarily, these supports provide an adequate frequency (goal of >50 Hz) for the sandwich equipment panel as well as the total module.

Each of the similar modules has the same primary structural elements and transfers the loads in the same manner. The power module was used for sizing and analysis because it has the highest equipment weight. The other modules are structurally adequate because they carry less weight. The actuation module includes hydrazine and nitrogen systems, and is configured to occupy available volume within the spacecraft structure. The solar array module, which includes the solar array drive assembly and associated equipment, is located above the transition ring because of the location of the array itself. This module uses the same attachment mechanisms as the other subsystem modules.

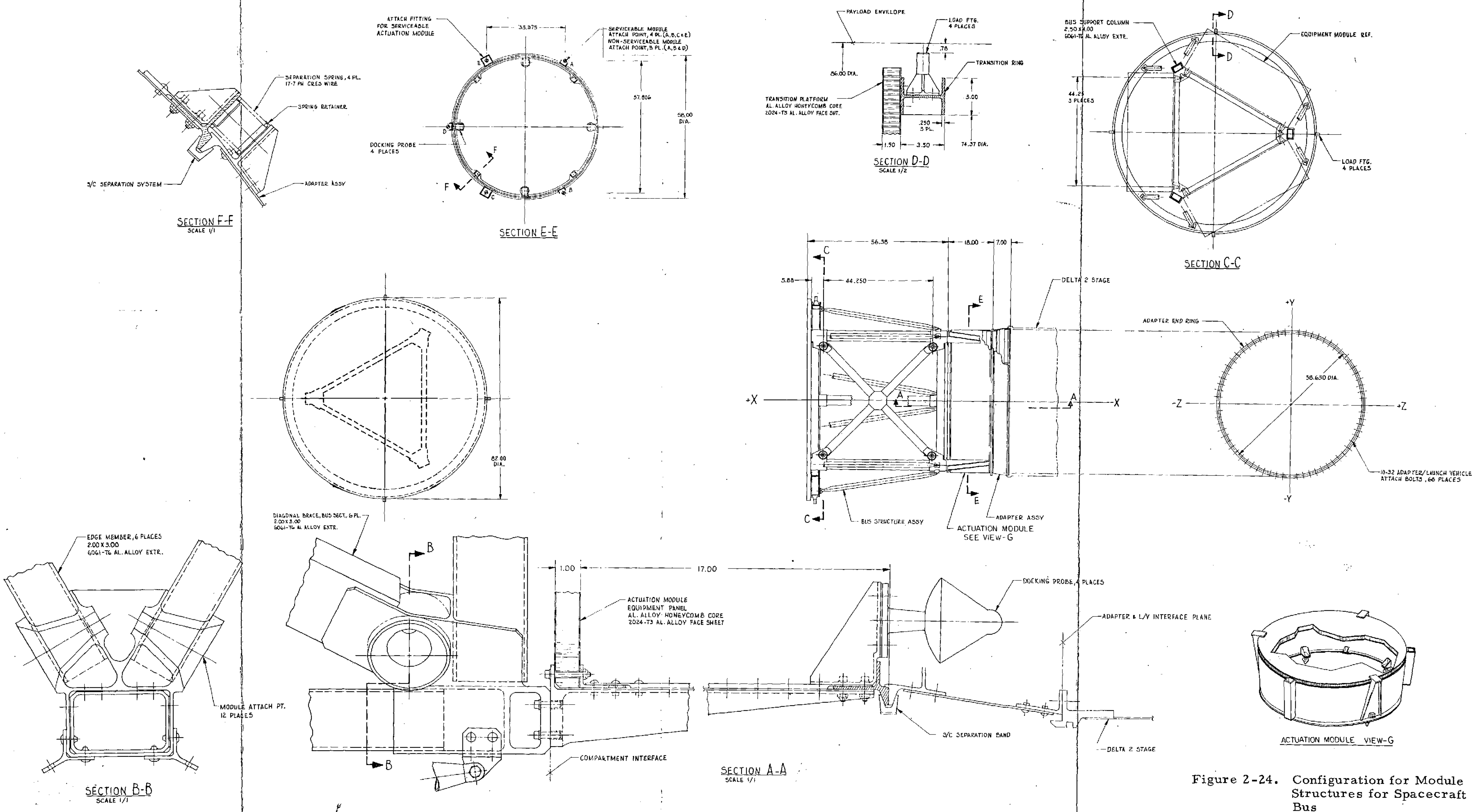

Figure 2-24. Configuration for Module Structures for Spacecraft Bus

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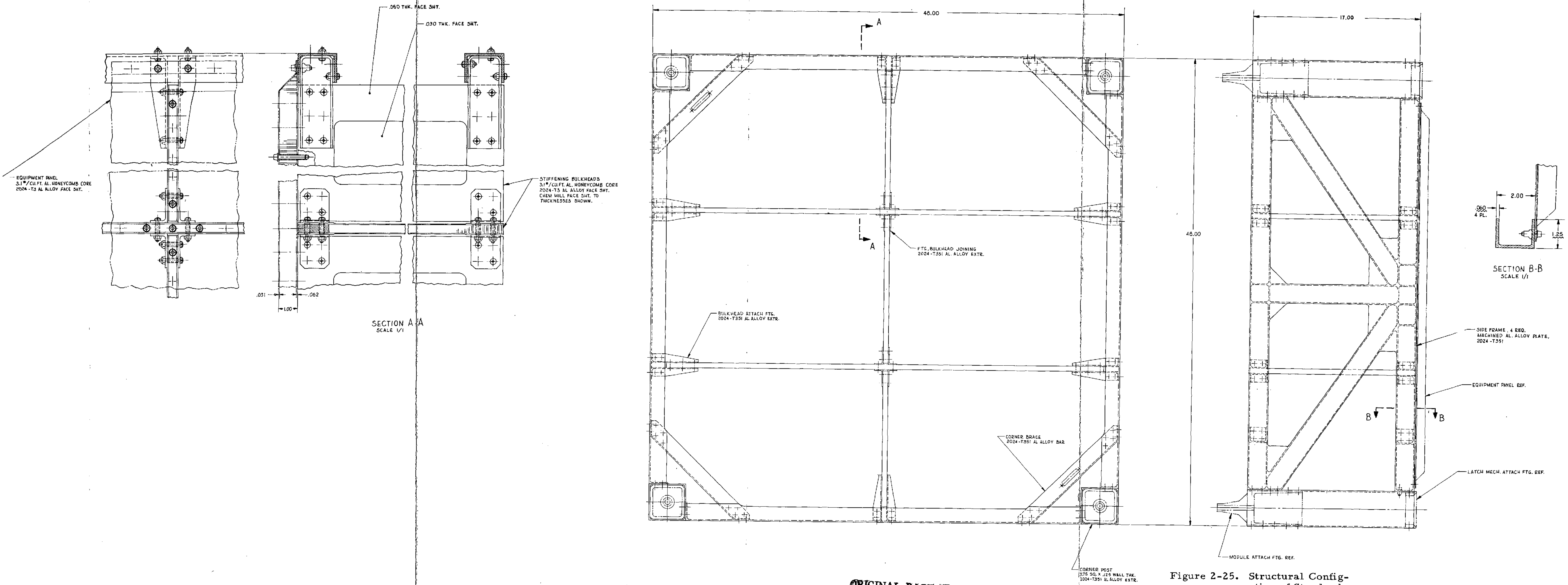

Figure 2-25. Structural Configuration of Standard Bus

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Our design requires that individual modules be essentially independent thermally. This provides design flexibility by allowing replacement and/or substitution of modules without thermally impacting overall observatory thermal control design. It also reduces test costs by obviating acceptance thermal-vacuum tests.

Conductive isolation is achieved by controlling mechanical module-to-structure coupling and temperature gradient across the interface. Multilayer insulation (MLI) between module and structure minimizes radiative interaction.

Each module has its own independent thermal control system consisting of a temperature-controlled heater and high-efficiency multilayer insulation blankets over all surfaces except for the radiating area, which has a low α_s/ϵ coating.

The payload and subsystem structural frames (including the transition ring) have independent thermal control provisions consisting of multilayer insulation blankets, sandwiching between structural members, and several independently-controlled heaters. Temperature level, distribution, and fluctuations are constrained to limit thermal distortion.

3. PAYLOAD DATA HANDLING

A survey of payload data handling requirements for medium-weight earth-orbiting missions yields the data shown in Table 3-1. As can be seen, a variety of formats exist and rates range from a few kbit/sec for certain instruments to several hundred Mbit/sec for some combinations of instruments. In addition, it may be desirable to transmit selected subsets of data in real-time to local users having receiving stations near regions of their interest. Typically, these local users have tight budgets. Thus, on-board processing, which yields transmission rates and formats that are easily manageable by such users, are likely to be quite cost-effective. A principal output from our study was the identification of value in modularizing payload data handling, both on-board and on the ground.

3.1 ON-BOARD DATA FORMATTING

Analyses of the data handling requirements presented in Table 3-1 indicate that most missions can be handled by common on-board equipment if these capabilities were available:

- Variable data rates
- Variable word sizes
- Variable number of inputs multiplexed
- Variety of framing patterns.

To make this possible, TRW recommends the development of a modularized data handling system (MODS) that can accept information from multiple channels, convert from analog-to-digital form if required (with quantization up to 10 bit/sample), and provide a single-channel serial bit stream of predefined but flexible format by multiplexing individual channels. Such a system has been defined during this study. It yields output bit rates in multiples of 2 to 120 Mbit/sec. Higher rates can be accommodated by operating several systems in parallel, and either phase multiplexing onto a single carrier or separately modulating into several non-overlapping bandwidths.

Table 3-1. Payload Data Handling Requirements

Mission	Number of Instruments	Number of Channels	Data Format A/B/C*	Total Data Rate (Mbit/sec)	Output Products
EOS-A	2	126	~2640/400/8 (TM) ~1200/150/6 (MSS)	135	Images CCT's
EOS-B	2	404	~2640/400/8 (TM) ~ 64/304/8 (HRPI)	240	Images CCT's
SEASAT	4	14	1000/1/8	10	CCT's Maps Data Printouts
SEOS	4	1925	2000/1/8	70	Images Data Printouts
SMM	5	60	128/128/8	0.005	Data Printouts

* A = Major Frame Size (Number of Minor Frames)
B = Minor Frame Size (Number of Words)
C = Word Size (Number of Bits)

The MODS system consists of a controller (physically separate from payload elements) and encoders (one of which is included in each instrument). Each encoder employs one or more multiplexer slices to gather the data from the multiple inputs (up to 64 per slice) and A/D converters to yield a serial digital output data sequence. Each encoder is directed by the controller when to gather data from each channel. Flexibility is obtained through the use of multiple encoders and the sequence in which the controller calls on the inputs. When used to achieve data rates higher than 120 Mbits, multiple controllers can be synchronized by a common clock. Mounting the encoders with the instruments shortens leads for ease in control of timing and noise. The encoders can issue the bits of each encoded word in parallel as well as serial form, facilitating data compaction for transmission to low-cost ground stations (LCGS).

Data are formatted by the controller into major and minor frames. Major frames can have up to 8192 minor frames, which in turn can have up to 512 words. Word length can also be programmed for individual instruments. The multiplexing sequence within both major and minor frames is controlled by read-only memories (ROM's) or programmable read-only memories (PROM's) peculiar to each instrument.

The system organization of the MODS multiplexer is shown in Figure 3-1. Modular building blocks having 64 analog inputs each can be assembled in various numbers to meet mission requirements. The analog multiplexer is a two-tier system to reduce capacitance and errors from current leakage through switches in the common output mode. A serial-parallel feedband A/D converter achieves the required accuracy at rates to 120 Mbit/sec.

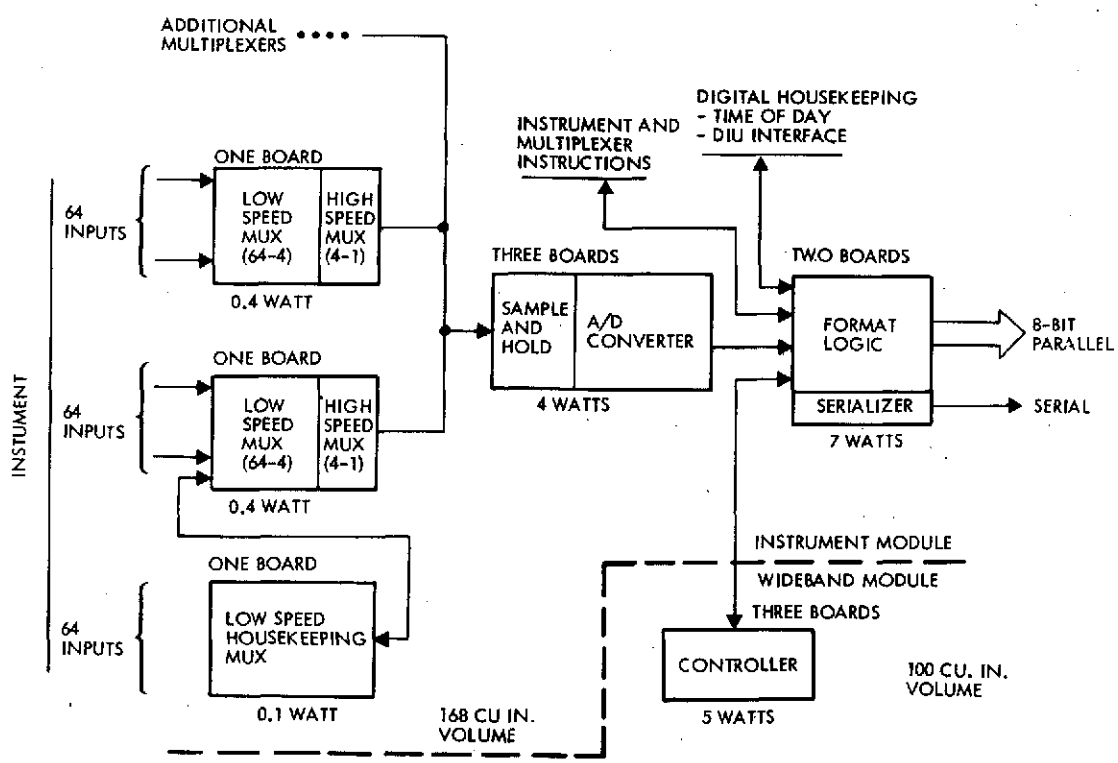

Figure 3-1. MODS Multiplexer System Organization

A wideband tape recorder can be incorporated into the data handling complement. The video recorder developed for ERTS yields combined data rates to 15 Mbit/sec and is a good candidate. Data rates of 120 Mbit/sec require development of a new tape recorder. Fortunately, the modular approach permits the incorporation of tape recorders or the capability for real-time transmission through TDRSS (see Section 3. 2. 3).

In some applications it may be desirable and feasible to transmit certain data directly to users, bypassing the normal system collections and central data processing functions. Cloud images sent via APT from Tiros satellites are an example. For higher resolution earth observation images, such transmissions become substantially more difficult. For example, instruments may collect data in several spectral bands, generate scan patterns with excessive retrace times, scan in an alternate forward and reverse mode, or generate conical scan patterns. Reducing the data rate to a more manageable level and converting these scan patterns to easily managed data streams calls for a speed buffer. Once implemented, it can provide a selection of data formats for local users.

The data buffer offers 6 to 1 data compaction (from 120 to 20 Mbit/sec) for a high-resolution multispectral scanning thematic mapper (TM). In addition to the rate reduction, our implementation of the speed buffer

- Accepts a complex interweaving of spectral bands and successive scanned lines and emits a data stream sequenced to be most useful to local users,
- Closes retrace gaps or reverses alternate direction scans as needed, and
- Permits selection of various combinations of resolution, spectral band, and swathwidth and location.

The speed buffer has two major sections: the line stripper and formatter. The line stripper edits 120 Mbit/sec data to reduce the average data rate to 20 Mbit/sec, and the formatter speed buffers and reformats the data to line sequential.

A block diagram of the speed buffer is shown in Figure 3-2. The editing instruction to the line stripper is received by ground command through the digital interface unit (DIU). The reformatting instructions are contained in the format memory ROM. Editing changes from the baseline can be made by changing the format memory and adding (or subtracting) data memory slices/boards.

3.2 PAYLOAD DATA TRANSMISSION

Payload data may be transmitted to dedicated NASA facilities or directly to various local users equipped with the proper receiving, recording, and/or processing equipment. The latter case requires attention to data formats, radiated power, and antenna pointing requirements to ensure low-cost implementation. Transmission to NASA facilities can be via dedicated STDN station or TDRSS.

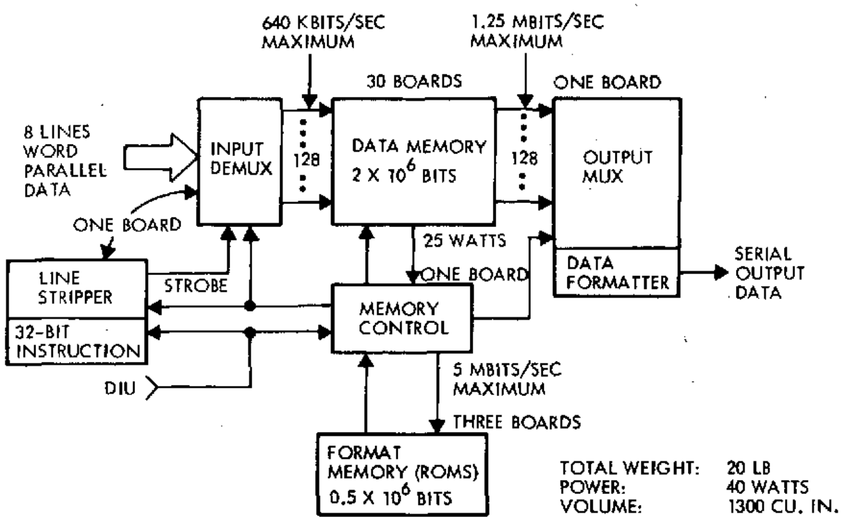

Figure 3-2. Speed Buffer (LCGS) Partioning (Includes a Line Stripper and Formatting)

3.2.1 Transmission to STDN Stations

Combined payload data rates less than 500 kbit/sec are most economically sent via the USB system employed for spacecraft telemetry and commanding. Data can be collected at any of the STDN stations equipped with USB equipment. Higher data rate payloads require mission-peculiar communication equipment. Here STD stations may be selectively eliminated, depending on their location and their ability to receive, synchronize, and record the transmitted signal. For the highest data rates, only the ERTS data reception stations (Alaska, Goldstone, and NTTF at GSFC) need be considered, and even here modifications and augmentations may be necessary or desirable.

Carrier frequency choice can be limited to S-, X-, or Ku-band. Considering that (1) S-band can accommodate only limited bandwidth, (2) NASA currently has no authorization to employ X-band for experimental transmissions, (3) NASA plans to incorporate a Ku-band capability at STDN stations, and (4) Ku-band is compatible with TDRSS;

selection of Ku-band for future wideband payload data appears sound. A survey of equipment technology at Ku-band indicates that at the power levels required, the cost differential between X- and Ku-band is small.

For any choice of frequency, a pointable antenna onboard the spacecraft is called for at the higher data rates. The ERTS data rate of about 15 Mbit/sec appears to be the upper limit for which nonpointable antennas makes sense. A 500 mW transmitter operating into a 2-foot pointable parabolic dish provides adequate performance with easily achievable and low-cost on-board equipment. Two such antennas and transmitters are needed for redundancy and for simultaneous transmissions to both the dedicated facilities and low-cost user terminals. Figure 3-3 illustrates a typical wideband communications and data handling module usable for many missions.

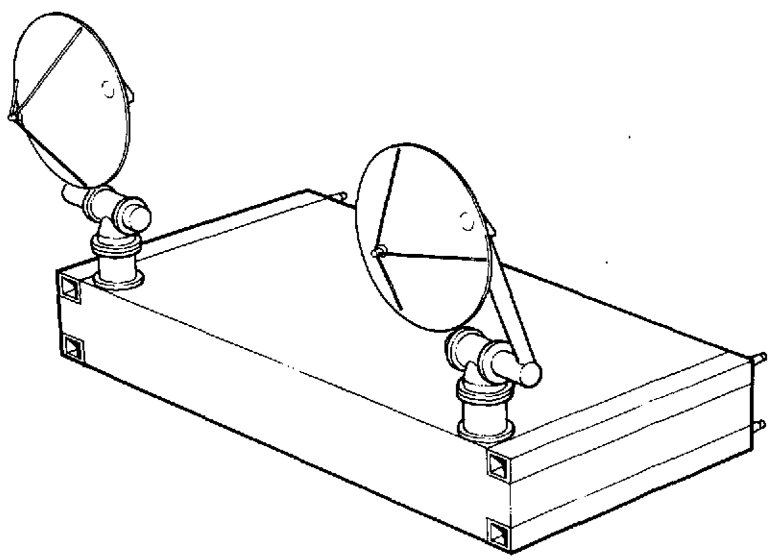

Figure 3-3. Wideband Communications Module

3.2.2 Transmission to Local User Stations

The design of low-cost local-user stations to receive directly transmitted data (e.g., the APT of the Tiros system) requires consideration of the information options available; the number of users desiring the service; the relations between data rate, cost, and data usefulness; and data formats that minimize ground equipment cost and complexity.

For earth observation missions, meaningful data are received only if data rates are at least 10 to 20 Mbit/sec. At these rates, a broadcast type transmission focusing energy over a limited area directly below the satellite is entirely feasible and requires about 10 watts of transmitted power. To receive such transmissions, ground stations require approximately 10-foot diameter antennas and low-noise preamplifiers. Alternately, the same ground stations can receive directed transmissions from the satellite using the transmitter/antenna combination described in Section 3.2.1.

Our studies indicate that both approaches are useful: the broadcast mode can be reserved for a standard data compaction format (see Section 3.1.2) valuable to all users (perhaps a single image, combining several spectral bands into their first principal components); the directed beam (which requires a priori scheduling and adjudication among competing users) can transmit in any of the spectral-band, swathwidth and location, and resolution choices available. The modular approach permits the incorporation or deletion of the entire low-cost user option or any of its modes with little impact on the remaining system.

3.2.3 Transmission to TDRS

Payload data can be acquired over foreign regions out of sight of appropriately equipped STDN receiving stations. This is done using tape recorders in a store-and-forward mode or by transmission to either of two TDRS synchronously placed above the earth. The TDRS approach eliminates the delay inherent in tape recorded data and the costly development of a wideband tape recorder. Its disadvantages are a small blind region encompassing part of India from which neither TDRS satellite can be seen, and the significant range losses that must be overcome by large antennas and high transmitter powers. Achieving the viewing angles to benefit from full TDRS coverage also poses some satellite configuration difficulties.

Our design makes the TDRS capability mission-peculiar, incorporated only when mission requirements call for it. To handle data rates as high as 240 Mbit/sec, our design employs a 20-watt travelling wave tube (TWT) operating at Ku-band together with a 6-foot pointable antenna. Link power budget calculations indicate a 6 dB margin at a signal-to-noise ratio (SNR) adequate for 10^{-5} bit error rate (see Table 3-2).

Table 3-2. Ku-Band Link to TDRSS-RF Link Calculation

PARAMETER	NOMINAL VALUE	NOTES
TRANSMITTER RF POWER LEVEL (dBM)	43.0	20 WATTS
PRETRANSMISSION FILTER LOSS (dB)	1.0	---
RF WAVEGUIDE LOSSES (dB)	1.0	---
ANTENNA GAIN (dB)	46.7	6 FT DISH
ANTENNA POINTING LOSS (dB)	0.3	±0.14 DEGREE RSS POINTING ERROR
SPACE LOSS (dB)	208.5	42,800 km RANGE AT EDGE OF EARTH
TDRS ANTENNA GAIN (dB)	52.6	"TDRS USER'S GUIDE", X-805-74-176, p.A-15
TOTAL RECEIVED POWER AT OUTPUT OF TDRS ANTENNA (dBM)	-68.5	---
TDRS SYSTEM NOISE TEMPERATURE (°K)	710.0	"TDRS USER's GUIDE", X-805-74-176, p.A-15
TDRS SYSTEM NOISE SPECTRAL DENSITY LEVEL (dBM/Hz)	-170.1	---
TDRS RECEIVED SIGNAL TO NOISE SPECTRAL DENSITY RATIO (dB/Hz)	101.6	---
DATA RATE (dB/Hz)	83.8	240 MBIT/SEC QPSK DATA
RECEIVED ENERGY TO NOISE SPECTRAL DENSITY RATIO (dB)	17.8	---
TDRSS TRANSPONDER LOSS (dB)	2.0	"TDRSS USER'S GUIDE" X-805-74-176, p.A-15
DEMODULATOR LOSS (dB)	1.5	"TDRSS USER'S GUIDE" X-805-74-176, p.A-15
ENERGY TO NOISE SPECTRAL DENSITY RATIO REQUIRED FOR 10^{-5} BER	9.5	COHERENT DETECTION OF QPSK DATA
SYSTEM PERFORMANCE MARGIN (dB)	4.8	---

The spacecraft antenna is mounted on a deployable boom on the anti-earth side of the observatory. Its height above the satellite is sufficient to see the horizon without interference from the satellite in the fore, aft, and anti-sun sides. On the sun side the solar array may cause occasional blockage. The two degree-of-freedom gimbal is mounted with its outer axis parallel to the spacecraft pitch axis. Gimbal lock and outer gimbal stops are reached only in the vicinity of the earth's poles where both TDRS's are simultaneously visible and where data collection is at a minimum because of low-sun angles on the earth's surface. Figure 3-4 presents a plan view of the TDRS antenna in conjunction with a typical payload assembly.

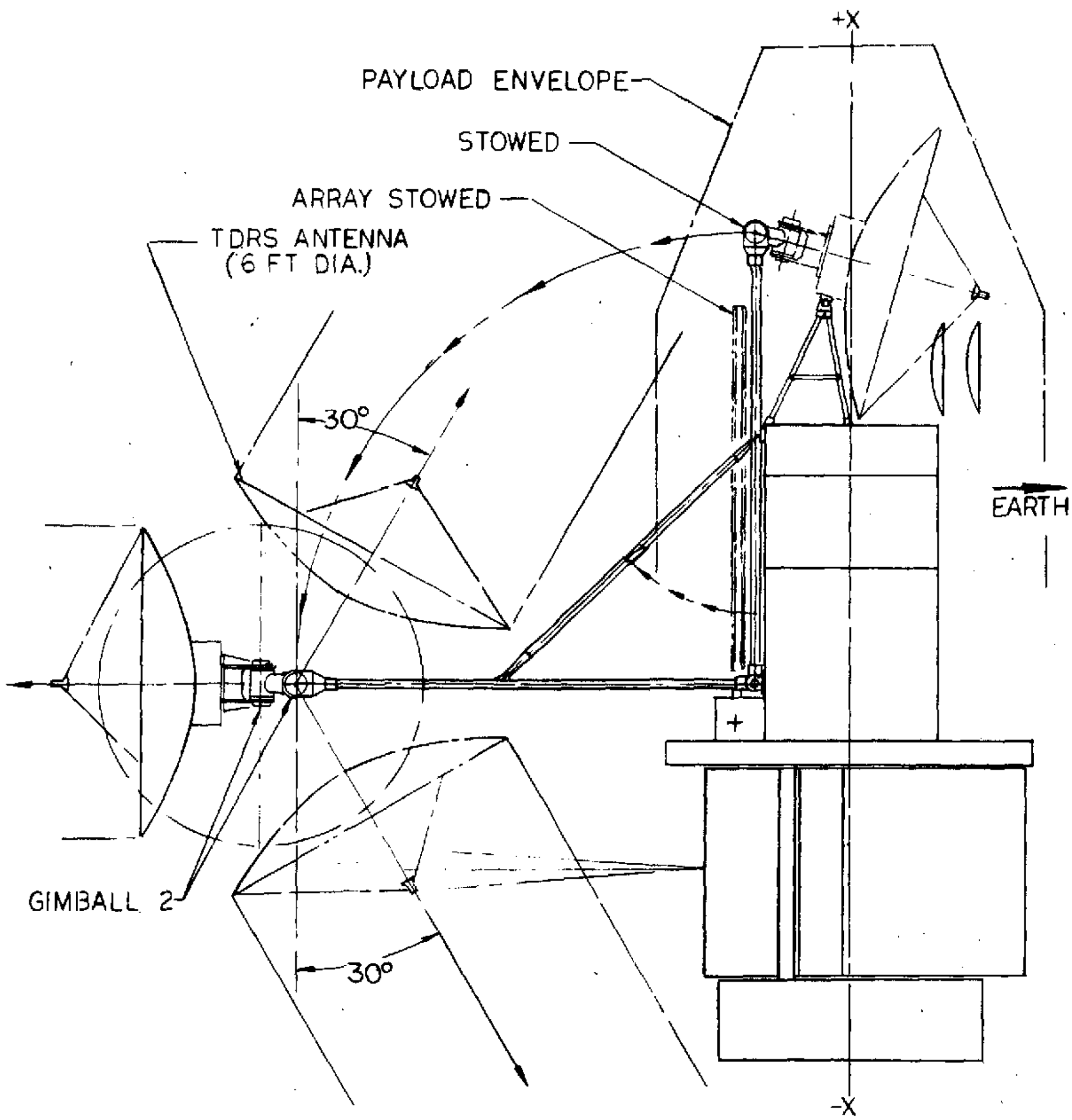

Figure 3-4. TDRS Antenna

3.3 STDN RECEPTION OF PAYLOAD DATA

Payload data from the observatory may be transmitted at X-band or at Ku-band, depending on factors outside the scope of this study. In either event, antennas at least 30 feet in diameter are required for most missions, and these will require new feeds and low-noise preamplifiers at the selected frequency. Because NASA plans to upgrade STDN stations to Ku-band, selection of that band for payload data reception appears most cost effective.

By constraining transmitted data rates to multiples and sub-multiples of two (7.5, 15, 30, 60, 120, 240 Mbit/sec), receiver design can incorporate standard selectable bandwidths and thereby be useful over the range of missions anticipated. Similarly, data synchronizers and tape recorders are most easily designed to operate at rates that are spaced apart by factors of two.

Our studies indicate that the ground station equipment should be transparent to format varieties. That is, the system sees a serial data stream arriving and delivers, on playback, a similar data stream without concern for word and frame sizes. Establishing word and frame synchronization then becomes a task for the data processing center. Bit synchronization, however, has to be established prior to recording, and if multiple-phase modulation is incorporated, data have to be demodulated and reserialized (or split off to separate tape decks) before recording.

Survey of tape recorder techniques indicates that only multitrack recording meets the demanding requirements of high-packing density, low-bit error rate and variable speeds to 120 Mbit/sec. Although no off-the-shelf unit has the required characteristics, the Ampex 2000 series recorders, which employ 42 parallel tracks on a 1-inch tape, come sufficiently close. The transport uses Miller coding at 18.5 kbit/inch/track. It runs at 180 inch/sec while recording or playing back at 120 Mbit/sec. Tape density is 660,000 bit/inch after deleting error detection bits, permitting 7.6×10^{10} bits on 9,600 feet of tape per reel. Record time at 120 Mbit/sec is 10-2/3 minute/reel. The Ampex system employs special synchronizing equipment to eliminate the effects of tape skew. A similar 2000 series system has been operating at acceptable performance at 80 Mbit/sec for over a year. Thus, the outlook for meeting the required performance at low development and recurring costs is excellent. These design features will enable the ground station equipment to be used without augmentation or alteration over the full range of expected missions.

3.4 CENTRAL DATA PROCESSING FACILITY

Although a central data processing facility (CDPF) for all missions of the modular observatory is attractive, NASA is creating facilities (STADAC and TELOPS) for handling low-to-medium rate payload outputs. Thus, the CDPF discussions following are applicable to the high-data rates, and to the formatting, correction and calibration of imagery, and particularly multispectral imagery of the earth's surface. Radar and passive microwave images may require special-processing equipment in addition to that discussed. The functional structure for a typical facility is shown in Figure 3-5.

Keys to the success of a modular CDPF are well-thought-out and defined interfaces, just as for the observatory. Here the interfaces of concern are the type of media and formats and the degree of data refinement for data inputs to the system, delivery to users, and archival data. Modularity is achieved best by organizing the system around four distinct operations:

1) Data preparation

2) Data refinement

3) Output product generation

4) System management.

This organization is shown in Figure 3-6.

In practical systems, information is lost each time collected data are altered by calibration or geometric manipulation. Although TRW's cubic convolution process minimizes these losses, cascaded application even here causes needless degradation. This problem is overcome by doing the entire correction in one step after the various correction

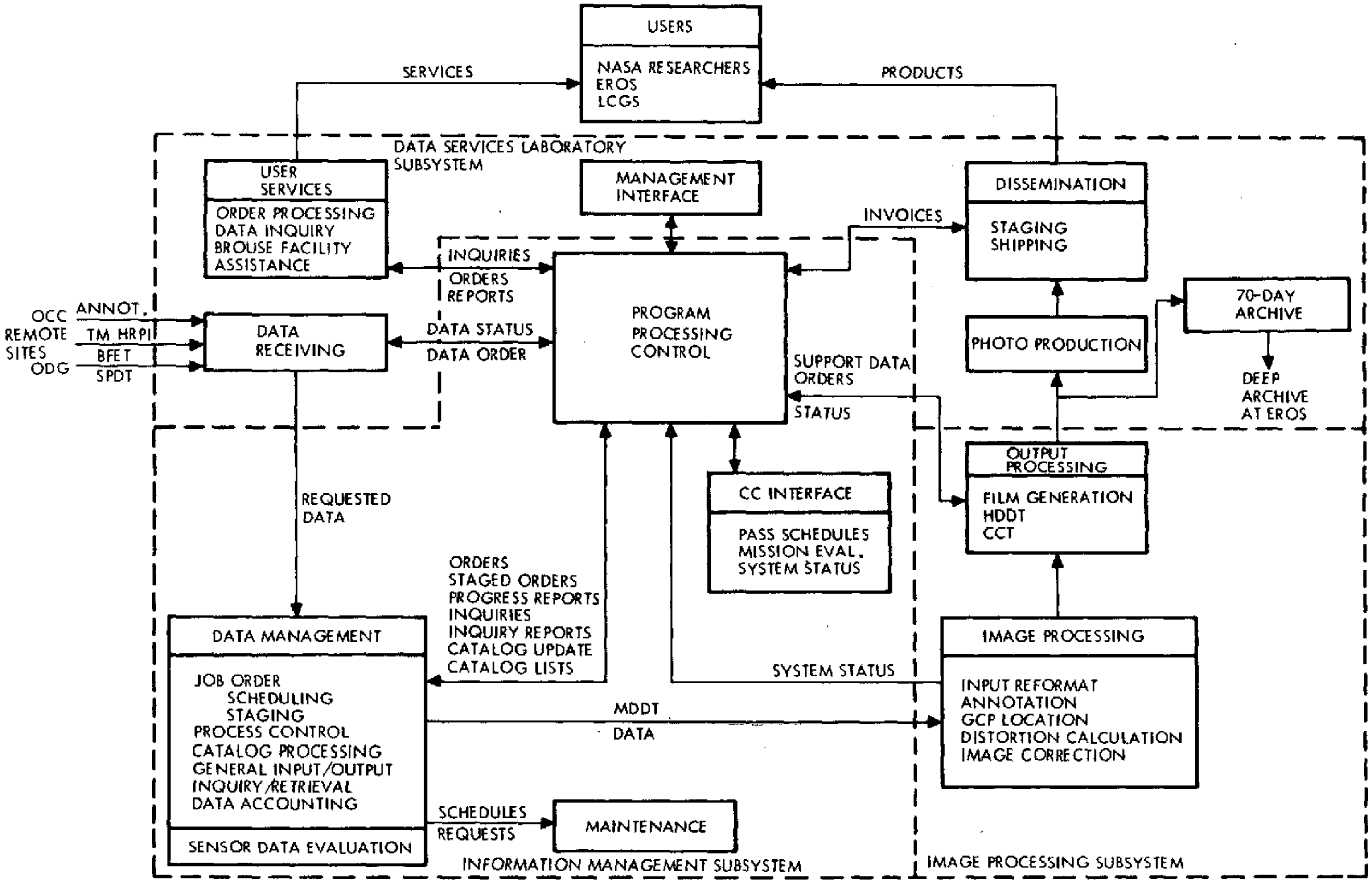

Figure 3-5. Central Data Processing Facility Functional Structure

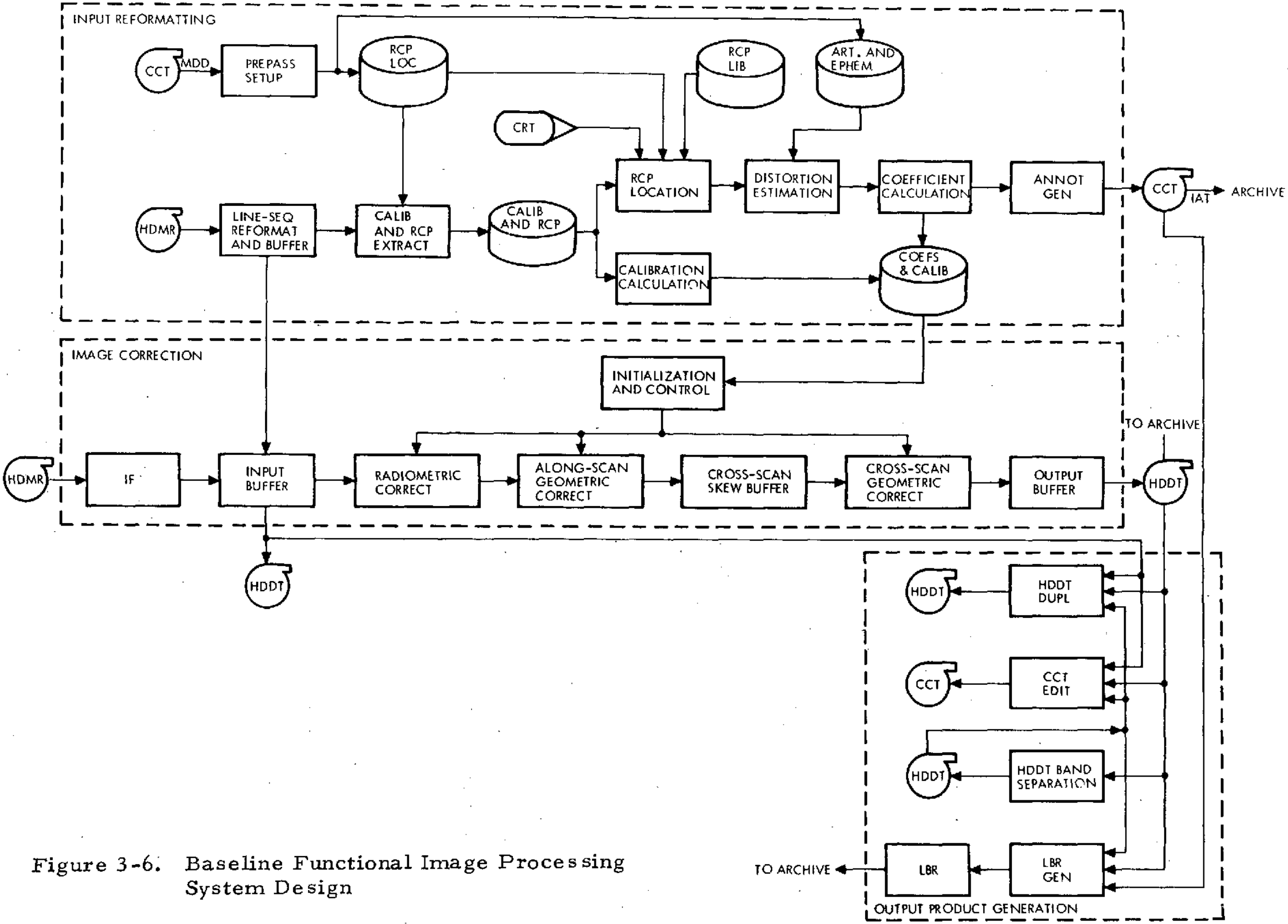

Figure 3-6. Baseline Functional Image Processing System Design

instructions have been combined, rather than several steps (i.e., a separate correction for each error source or map projection). This poses questions regarding steps common for all users versus those peculiar to certain users, such as gamma correction and map projection.

The degree of correction for archived data raises similar questions. Because high-speed techniques are available for geometric and photometric manipulation of image data, they are no longer bottlenecks in the final reproduction process, and image corrections can proceed as rapidly as output media can record them. TRW therefore recommends that the archive copy of all images be completely uncorrected, accompanied by instructions for calibrating and correcting them to some standard. Users desiring alternate outputs need only modify these standard instructions and apply the resulting ones to the original data to obtain a highest quality first-generation copy of the image in the desired format. Thus, after calibration and ground control point (GCP) regions are abstracted in the data preprocessing, and after calculations based on these abstracted data have been completed, the resultant instructions must be merged with the original data for archiving and for passing on to the image correction process.

3.4.1 Data Preparation

The standard input to the CDPF is magnetic tape recorded on a high-density multitrack recorder (HDMR). A similar machine (or multiple machines if required for throughput) is incorporated at the CDPF for playback.

To prepare the data for correction or to generate an output product may require any or all of the following functions:

- Synchronize word, minor frame, and major frame
- Separate and reorder data from different spectral bands or different detectors operating in parallel (for the TM, data from 16 detectors collected in parallel must be converted to line serial)
- Reverse data in alternate frames to correct for alternate front and back mirror scans (Hughes TM)
- Abstract selected calibration data, timing, and/or regions containing GCP's

- Locate coordinates of GCP's in abstracted regions
- Calculate correction coefficients that define transformations from received data to corrected data.

These functions can be performed in a medium-capacity computer system (such as Xerox 550) supported by a special-purpose hardware buffer and reformatter. This special-purpose unit serves as an interface between HDMR and the computer. It includes minor and major frame word synchronization circuits and enough memory to double buffer a full major frame of data. Thus, data entered in the order collected can be output in the sequence required for processing. The computer processes abstracted data to first yield registration control point (RCP) coordinates, then calibration and correction coefficients.

Data preparation begins with the first of two playbacks of a received HDMR tape. During this pass only abstracted data are retained. These data are used for five major tasks: RCP location, detector calibration, error estimation, distortion coefficient calculation, and annotation generation. Secondary tasks include maintenance of the RCP and calibration data library.

In the RCP location phase, abstracted areas are analyzed automatically for the precise location of the specified RCP within the scene. If the automatic RCP location processing fails with the preselected number of RCP's within a scene or their alternates, the operator may intervene to resolve the difficulty. The operation may select a new set of RCP's to avoid local cloud cover not predicted by the meteorological planning information. Reference RCP's are stored in the RCP library with a 100 Mbyte disc file. The results of the RCP location process are passed to the error estimation phase.

Detector recalibration occurs only infrequently; the abstracted calibration data are normally scanned only for limit variations. If repetitive limit variations are found, a recalibration is called for and the abstracted calibration are processed completely and used to update the data base for that instrument.

The error estimation task calculates the sensor pointing vector as a function of time, using best-fit ephemeris and RCP location measurements. A sequential estimator is used to estimate the sensor pass dynamics. The updated pointing information is passed to the distortion coefficient calculation process; here are generated the image correction parameters necessary to map the distorted input sensor data into the corrected output coordinate system.

Annotation files are constructed for each pass/scene to supply data identification and labeling. Included are data generated during image data processing, such as frame time and number and nadir point, as well as special annotation provided by the information management system.

The header data and annotation files, including the calibration data, are placed on a cartridge disc pack, and the pack is stored with the HDMR tape for image correction processing.

3.4.2 Data Refinement

After data preparation, a second reading of the HDMR tape produces data at the output of the buffer/reformatter unit that are ready for processing. Refinement takes place in a special-purpose interpolator operating under control of a medium-capacity computer system. The data are corrected for radiometric calibration errors and are transformed geometrically to remove system errors and to format the output data for map projection so that geometric displacement from a predefined grid is never greater than 1/4 pixel.

The resultant data stream is recorded on a high-density digital tape (HDDT) where spectral bands are interleaved pixel by pixel. In parallel with this correction process, an uncorrected HDDT is also generated. This tape contains data in a fully line-sequential, pixel-interleaved format in which the calculated correction coefficients are inserted in the retrace interval between major frames. It is this uncorrected tape which is archived and used to generate specially processed or alternate map-projection outputs.

A process control task within the computer has as its primary function the management of the various steps in the image correction process. Its data management functions include control of the HDMR subsystem, routing of the radiometric calibration data, management of the distortion correction coefficients used by the geometric correction hardware, and control of the HDDT generation for primary output products.

The special-purpose interpolation hardware performs a cubic convolution resampling* of the input data in two dimensions to yield fully corrected output data. It is organized into three basic sections: an along-scan interpolator, an intermediate skew buffer, and an across-scan interpolator.

The along-scan correction hardware performs radiometric calibration of the input data using data supplied from the calibration data base. Then calibrated data are corrected in the along-scan direction using the distortion coefficients from the distortion calculation function. The output of the along-scan corrector is placed into a large core skew buffer for use by the cross-scan correction hardware.

The along-scan corrected data are read from the skew buffer by the cross-scan correction hardware as sequential input lines to produce continuous corrected output lines. The output of the cross-scan hardware is passed directly to the output HDDT system to create the corrected image tapes.

3.4.3 Output Product Generation

Three HDDT's are the principal products delivered to EROS at Sioux Falls: two HDDT's are part of the system output products; another is a corrected HDDT in which the continuous data stream is separated into overlapping scenes and organized so that spectral bands are image sequential rather than pixel sequential. From these three HDDT's, all photographic copies and computer-compatible tapes are generated. The output-product-generation function produces HDDT tapes for EROS and other products required by NASA principal investigators.

The output-product-generation system is built around a medium-capacity computer and several large-capacity (100 Mbyte) moving head

*See Section 7.9 of Appendix A to Report 3 of this series for a detailed discussion of the cubic convolution process and its advantages over other processes.

disks. Also included are several CCT recorders and a laser beam recorder for producing digital and photographic output products, respectively.

Users requiring computer-compatible tapes have two formats to select from. The nominal industry standard today in CCT's is based on 800 bit/inch NRZI and 1600 bit/inch PE tape technology. Over the next 2 to 5 years, tape technology is expected to include the 6250 GCR now entering commercial practice.

The CCT subsystem is a dual density 1600/6250 bit/inch PE/GCR drive and matching controller. The tape drive is dual speed as well as dual density. The highest speed, 250 in./sec, is used at 1600 bit/inch to yield a transfer rate of 400 kbit/sec and a tape copy time of less than 2 minute/2400 feet. The 125 in./sec speed is used at 6250 bit/inch to give a high transfer rate of 790 kbit/sec.

First-generation positive and negative transparencies with a 241-mm (9.5-inch) film format are generated from digital data and presented in a standard output coordinate system. The LBR that generates these has the following characteristics:

- Continuous film motion and rotating mirror scan
- 50 percent MTF at 40 lp/mm with f/32 optics
- 256 grey levels
- Up to 5×10^6 pixel/sec writing rate.

The same computer is used for each of the three processes (except for peripheral complement); depending upon throughput requirements, all three processes may be done in the same unit or multiple units may be used. Generation of the three standard HDDT outputs for up to 2×10^{11} bit/day is feasible. Production of multiple output products at this rate can easily force expansion of the system to three or more computers.

3.4.4 System Management

In addition to the payload data manipulation functions, it is necessary to maintain files on and control production in the system, to retrieve archived data, to process user requests, and to perform related functions. These are done in the system management function whose design closely parallels that used for ERTS.

3.5 LOW-COST GROUND STATIONS

The LCGS makes earth observation data available to users in a timely manner and in a format that allows processing to their unique needs. Implementation of the LCGS requires an RF equipment subsystem for acquiring sensor data transmitted from the satellite at 20 Mbit/sec and an image processing subsystem to produce output products.

In keeping with our desires for modularity, we have provided for two basic configurations: direct display or record and process. Each design requires the same acquisition equipment and operational interface between the user and NASA. The designs differ in the approach to the image processing subsystem. The direct display design produces only film output products, recording them in real time; it is the lower cost system. The record and process design records then processes the image data using a minicomputer-based system; it produces a range of output products tailored to user needs, and output data quality is commensurate with that of the CDPF. Figure 3-7 summarizes the functional aspects of each design.

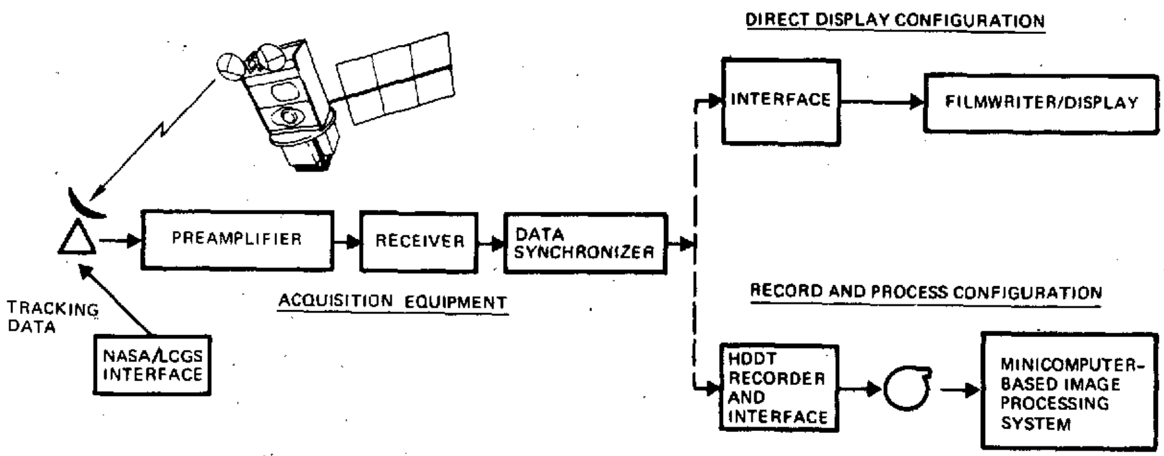

Figure 3-7. LCGS Functional Block Diagram

The quality of the LCGS output products differs for the two baseline designs. For both, radiometric calibration of the LCGS sensor data is performed onboard. Because the direct display design produces a film output product only, the radiometric accuracy of its output product is intrinsically less than the digital products of the record and process design. Furthermore, the direct display design performs no geometric corrections on the data. The film output products, then, have a geometric accuracy dictated by the inherent accuracy of the transmitted sensor data, which is equivalent to the accuracy of the uncorrected HDDT output product of the CDPF. As noted previously, the record and process design can produce output products with geometric accuracy comparable to that of the CDPF.

3.5.1 Acquisition Equipment

The acquisition equipment is the same in both designs; its function is to acquire the RF signal transmitted from the satellite. The most expensive component is the antenna system, consisting of an antenna, pedestal, feed, and control electronics. The pedestal is driven by a punched paper tape and is an existing design. Pointing information is prepared by the control center and punched at the LCGS using a telephone link and computer technology.

An uncooled parametric amplifier is used to help meet the SNR requirements for the communication link. The receiver incorporates a biphase demodulator. A bit synchronizer and decommutator complete the acquisition components.

3.5.2 Direct Display

The output of the acquisition equipment is a frame synchronized data stream. The direct display subsystem uses this data stream to produce an output film product. The data stream is buffered line-by-line then recorded on film using a continuous 9-1/2-inch laser beam film transport. The data are band-interleaved, i.e., line one of band one, then line one of band two, etc. The same line is scanned across the film, one band at a time, then the next scan presents the next sensor line. A maximum resolution of 8192 pixels per scan is required for the first three options, but the reduced resolution option requires 17,070 pixels for a laser scan line because of the seven bands. This is less than the 20,000 pixels attainable across a 9-1/2-inch film format for scanning mirror laser technology and is used to size the filmwriter resolution.

3.5.3 Record and Process

If improved performance is required, the sensor data must be recorded and additional processing performed. Figure 3-8 shows the proposed configuration and its output products.

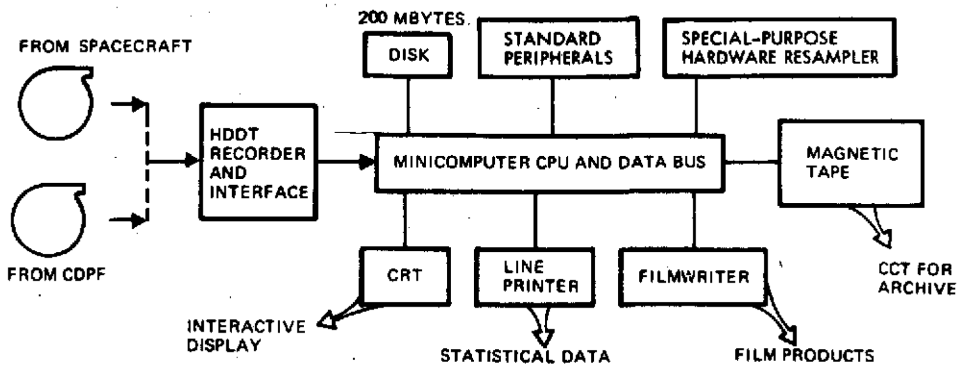

Figure 3-8. LCGS Minimal Cost Record and Process Configuration

The technology used for the HDDT recording at the CDPF is also used as the input tape recorder for the LCGS. This is a cost-effective approach to acquiring the LCGS input tape recorder and provides a tape format interface between the CDPF and LCGS. Should desired data be missed because of the on-board compaction mode selected or preemption of the transmission capability by a higher priority user, it can be recovered (after a delay) by request from the CDPF.

Note that there is a special-purpose hardware component in the record and process configuration that provides increased throughput, if desired. It is essentially a microprocessor for implementing high-order image processing algorithms and is optional.

The image processing functional flow is identical to that of the CDPF with two exceptions: radiometric calibration is not required, and the predicted ephemeris provided with the control center tracking data is used for the distortion estimation process. The lack of more precise ephemeris data results in a greater registration control point search region, but two registration control points per scene are sufficent to remove the bias errors and estimate the attitude errors.

4. APPLICATION OF THE MODULAR DESIGN TO VARIOUS MISSIONS

The value of a modular design is in the ease and cost effectiveness with which it can be applied. This section shows show a system can be synthesized to accomplish three of the missions defined in Table 2-1 (EOS-A, EOS-B, SMM). Alterations to the standard bus required to perform these missions are minor and are summarized in Table 4-1. A weight budget for each appears in Table 4-2.

4.1 THE EOS-A MISSION

The EOS-A mission is planned for launch in 1979. It includes a thematic mapper and an MSS. It will be launched on a Delta 2910 booster and can later be retrieved by the Shuttle. Payload data can be transmitted through a TDRS or directly to STDN ground stations and to low-cost ground stations. Figure 4-1 illustrates such an observatory.

4.1.1 Payload Characteristics

The baseline thematic mapper selected by TRW for the EOS-A spacecraft is the linear image plane scanner. It scans the ground area below the spacecraft (Figure 4-2), collecting radiation in each of seven spectral bands, and develops analog signals whose magnitude is dependent on the radiation intensity in each band. These signals, along with appropriate synchronization and housekeeping signals, are processed by the MODS unit, which performs the multiplexing and analog-to-digital conversion prior to their transmission. As indicated in Figure 4-2, an array of detectors with their fields of view displaced in the along-track direction is swept in a cross-track direction to cover a 185-km wide area. Scan timing is adjusted so that adjacent scan swaths are contiguous as the spacecraft motion in orbit advances the scanned field in the along-track direction. Instrument characteristics are summarized in Table 4-3.

The multispectral scanner for the EOS mission is a modified version of the MSS flown on ERTS-A and -B (characteristics shown in Table 4-4). Figure 4-2 is equally applicable to the MSS though the scan technique for the MSS differs from the Te' thematic mapper and the number of along-track detectors is 6, rather than 16. Modifications include addition of a

Table 4-1. Standard Bus Applicability To Selected Missions

Item	EOS-A(1)	EOS-B(2)	SMM(3)
Communications and Data Handling Module (See Section 2.6)	Two 8 K memory modules(4)	Same as EOS-A	Same as EOS-A
Attitude Determination Module (See Section 2.7)	No fine sun sensor	Same as EOS-A	Fine sun sensor (or payload data)
Actuation Module (See Section 2.8)	• 50 lb hydrazine • 35 lb Nitrogen • 060 pitch wheels (1.5 ft-lb-lb; 7 in.-oz) • Magnetic moment = 120,000 pole-cm • Single level N_2 thruster (0.1 lb) • 3 lb hydrazine engines (2)	• 1115 lb hydrazine(5) • 70 lb Nitrogen • 060 pitch wheels (1.5 ft-lb-sec; 7 in.-oz) • Magnetic moment = 120,000 pole-cm • Dual-level N_2 thrusters (0.1/1.0 lb) • 50 lb hydrazine engines (2)	• No hydrazine • 34 lb Nitrogen • 060 yaw wheels (7.2 ft-lb-sec; 20 in.-oz) • Magnetic moment = 120,000 pole-cm • Single level N_2 thrusters (0.1 lb)
Power Module (See Section 2.9)	2 batteries; 40 amp-hr each	Same as EOS-A	Same as EOS-A
Array and Drive Module (See Section 2.10)	• 24 subpanels (40 watts each) • Rotating array with 15 deg bend	Same as EOS-A	• 16 subpanels (40 watt/panel) • Body-fixed array
Structure (See Section 2.11)	• Nonreplaceable modules • Aft adapter	• Replaceable modules • Aft adapter	Same as EOS-A

(1) 381 nmi sun synchronous; direct inject with 2910; retrieval "high" only

(2) 381 nmi sun synchronous; Shuttle launched low; resuppliable

(3) 300 nmi, 33 deg inclination; 2910 direct inject; expendable (or retrieval "high" only)

(4) Redundancy not included (see Table 2-2).

(5) During first launch from an expendable booster into final orbit, only 500 lb of hydrazine is needed.

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Table 4-2. Weight Summary

Item	Weight (lb) EOS-A	EOS-B	SMM
Spacecraft	1486	2170	1578
Electrical power module	379	400	379
Solar array and drive module	177	198	144
Communications and data handling module	152	173	170
Attitude determination module	195	216	195
Actuation module	264	844*	215
Spacecraft bus assembly	319	339	475
Payload	1023	1233	2043
Wideband communications and data handling module	166	187	—
TM module	413	434	—
MSS module	171	—	—
TDRS module	106	—	—
HRPI module	—	399	—
Solar maximum mission	—	—	1430
Payload bus assembly	167	213	613
Adapter	87	124	131
Total Observatory Without Contingency	2596	3527	3752

*This is the weight for initial insertion into final orbit using a conventional booster; 550 additional pounds of hydrazine are required to boost to the final orbit after Shuttle servicing.

fifth band for thermal measurements and minor modifications needed to operate at the lower altitude (705 km). The anticipated changes from the five-band MSS design are:

- Open scan mirror bumper positions
- Increase image-plane field-stop size
- Increase mirror scan frequency by retuning mirror

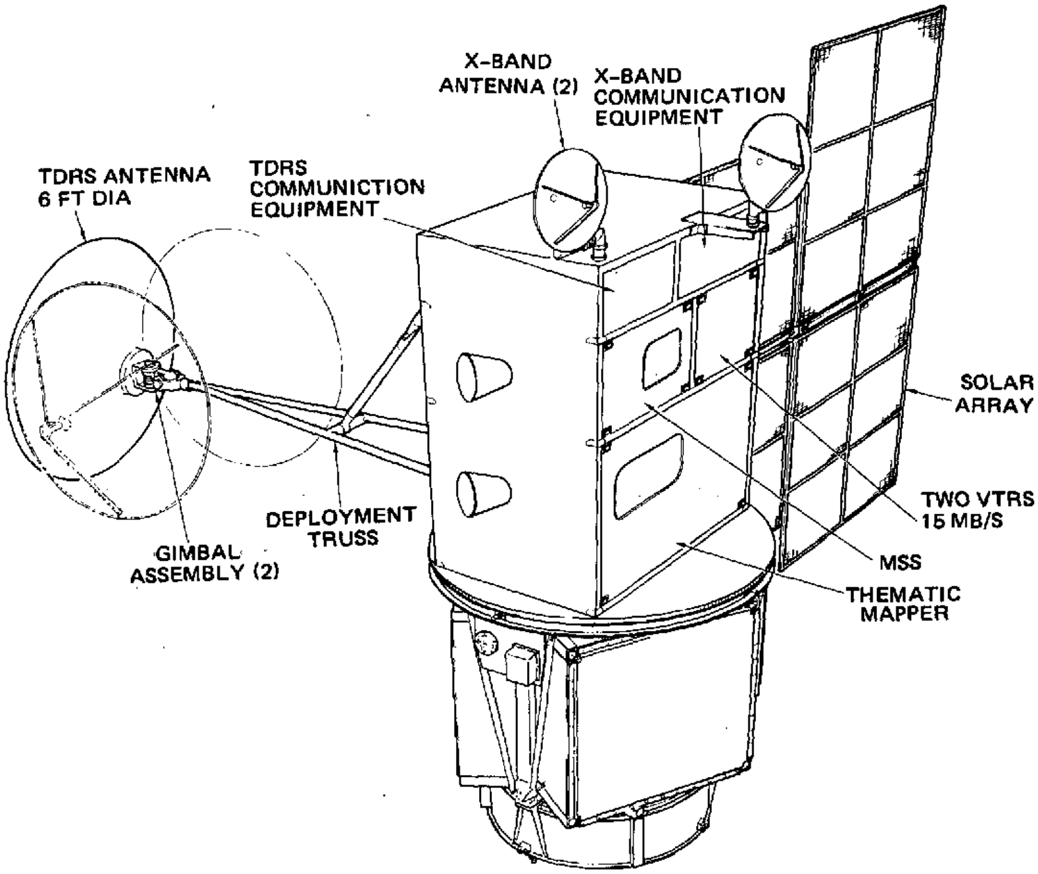

Figure 4-1. EOS-A with TDRS Antenna

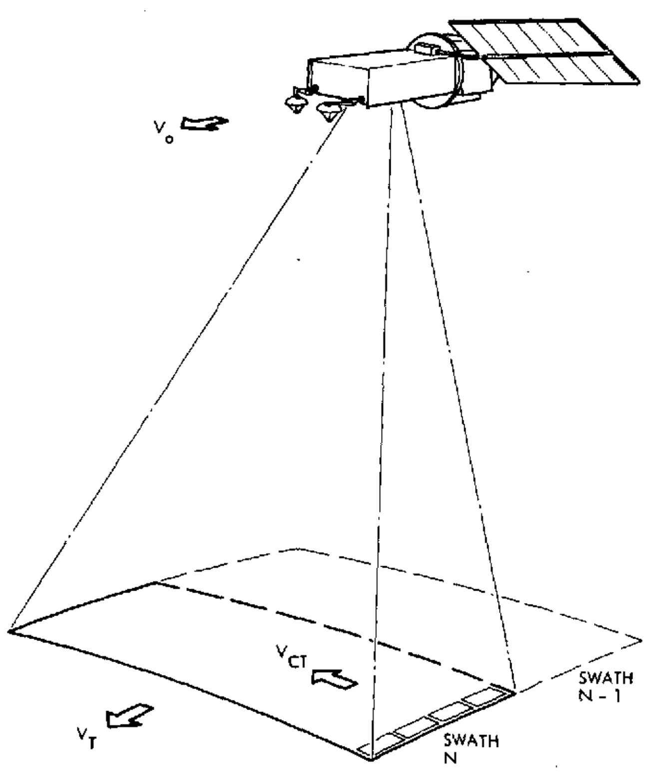

Figure 4-2.
Thematic Mapper
Scan Geometry

Table 4-3. Thematic Mapper Characteristics

Design altitude: 705 km

Ground resolution: 30 m

Swath width: 185 km

Sensitivity:

Spectral Band (μm)	Input Radiance (10^{-5} w/cm^2 sr)	S/N at Mininum Radiance
0.5 to 0.6	22 to 363	10
0.6 to 0.7	19 to 297	7
0.7 to 0.8	16 to 231	5
0.8 to 1.1	30 to 363	5
1.55 to 1.75	8 to 66	5
2.1 to 2.35	3 to 39	5
10.4 to 12.6	200 to 265	NEΔT = 0.5°K

Radiometric accuracy: ±10% absolute
± 1% relative (cell/cell and band/band)

Weight: 166 kg (365 lb)
Size: 183 x 97 x 102 cm (72 x 38 x 40 in.)
Power: 110 w
Output signals:

Source	No. of Channels	Data Rate words/sec/channel	Bits/word
Bands 1 through 6 data	96	115,165	8
Band 7 data	4	28,790	8
Housekeeping and Command Verification	50	0.5	8

Modulation transfer function: MTF >0.5 for a spatial input frequency of 30 m per half cycle

In addition, we recommend that the output be converted from 6- to 8-bit words to reduce the quantization noise. This will take advantage of the increased signal-to-noise arising from the lower altitude. The resulting increase in data rate from 15 to 20 Mbit/sec leads to compatibility with the compacted data sent to the LCGS.

Table 4-4. ERTS MSS Characteristics

Design Altitude:	915 km
Ground Resolution:	90 m
Swath Width:	185 km
Sensitivity:	

Spectral Band (m)	Signal Input for 4 V Output ($w/cm^2/sr$)
0.5 to 0.6	0.00248
0.6 to 0.7	0.00200
0.7 to 0.8	0.00176
0.8 to 1.1	0.00460

Weight:	50 kg
Size:	Approximately 36 x 38 x 89 cm
Power:	69W
Output Signals:	

Source	No. of Channels	Data Rate (words/sec/channel)	Bits/word
Signal Channels	24	104,000	6
Telemetry Channels	97		

These modifications will enable the revised MSS to scan a 185-km swath beneath the spacecraft with 90-meter resolution in five spectral bands at an improved signal-to-noise ratio. All data will be sent to the CDPF and requested data to the low-cost ground stations.

4.1.2 On-Board Payload Data Handling

The principal output of the MODS multiplexer in the TM is a serial bit stream at 120 Mbit/sec ready for modulation and transmission. On the other hand, the output of the MSS is 20 Mbit/sec and is synchronized to the clock used by the TM. Error rate is decreased for both the MSS and TM by implementing a second 120-Mbit/sec data stream consisting of a mixture of 1/3 MSS bits and 2/3 TM error-correction bits. The two data streams then are delivered to a quadriphase modulator to create an RF signal for transmission either directly to STDN stations or the CDPF via TDRSS.

The MODS multiplexer also outputs a word-parallel data stream to the high speed buffer. The EOS-A implementation of the general system accepts data from the MODS encoder in minor frames consisting of 16 detectors in six bands and four detectors in a seventh band, and converts them to any one of the following forms:

- All data from one band (out of seven possible), full swath
- Two bands for one-half swath
- Four bands for one-quarter swath
- Seven bands, reduced resolution (90 meters)

In the options that provide more than one spectral band, the pixel-sequential data stream is converted to line sequential so that a single ground-based filmwriter can lay down all spectral bands side by side, taking into account the reduced swathwidth in each band. Finally, prestored radiometric corrections are made to the data so that direct recording at the low-cost ground station yields an improved output product. On command, MSS data can be substituted for compacted TM data for transmission to the low-cost ground stations.

4.1.3 Standard Spacecraft Bus

The standard spacecraft bus is easily configured to perform the EOS-A mission (Table 4-1). To conserve weight and because it will be launched in the pre-Shuttle era, provision is made for recovery by Shuttle, but not for on-orbit servicing. In this configuration, an aft adapter is employed for launch, and the transition ring is retained for Shuttle rendezvous.

The communications and data handling module is configured in the baseline mode. The computer memory is sized at two 8000-word modules.

The two 40 amp-hr batteries in the power module supply up to 30 minutes of payload operation per orbit. Correspondingly, the solar array has been sized for 24 subpanels to maintain the battery charge for three years at maximum duty cycle. The solar array off-axis angle is 15 degrees.

In the EOS-A orbit, the attitude determination module provides an attitude reference accurate to 30 arc-sec and stable to better than 0.009 deg/hr. The actuation module controls spacecraft pointing accuracy to that established by the attitude determination module. Three OGO pitch wheels (1.5 ft-lb-sec), orthogonally placed, are employed. Magnetic torquers are sized at 120,000 pole-cm. The nitrogen supply for reaction control during acquisition and orbit adjust over a three-year period weighs 70 pounds.

An orbit-adjust capability consisting of 50 pounds of hydrazine and delivering 100 ft/sec adequately compensates for initial booster insertion errors and provides about three years of drag make-up at the designated altitude.

Table 4-1 summarizes these characteristics for the EOS-A configuration.

4.1.4 Ground Data Handling

The ground data handling system for EOS-A is the modular configuration described in Section 3. The buffer/reformatter hardware, which interfaces the HDMR with the computer, must be programmed to de-interleave the parallel scanned TM data. The 240-Mbit combined data stream is received on two separate tapes, one for each 120 Mbit component. Thus, the buffer/reformatter has to accept the error detecting bits on one tape while processing the other to obtain value from these additional bits. The number of computers and peripherals required can be limited to one for all functions if responsibility for supplying all but NASA output products is placed with EROS at Sioux Falls. Otherwise a second computer must be added.

Either LCGS option described in Section 3.5 can be used.

4.2 THE EOS-B MISSION

EOS-B is similar to EOS-A but incorporates a high resolution pointable image (HRPI) in place of the MSS. For a 1981 launch, it requires a Delta 3910 launch vehicle. The increased launch vehicle capability permits the inclusion of complete on-orbit serviceability or instrument upgrading. Figure 4-3 illustrates this observatory and Tables 4-1 and 4-2 convey significant configuration and mass properties.

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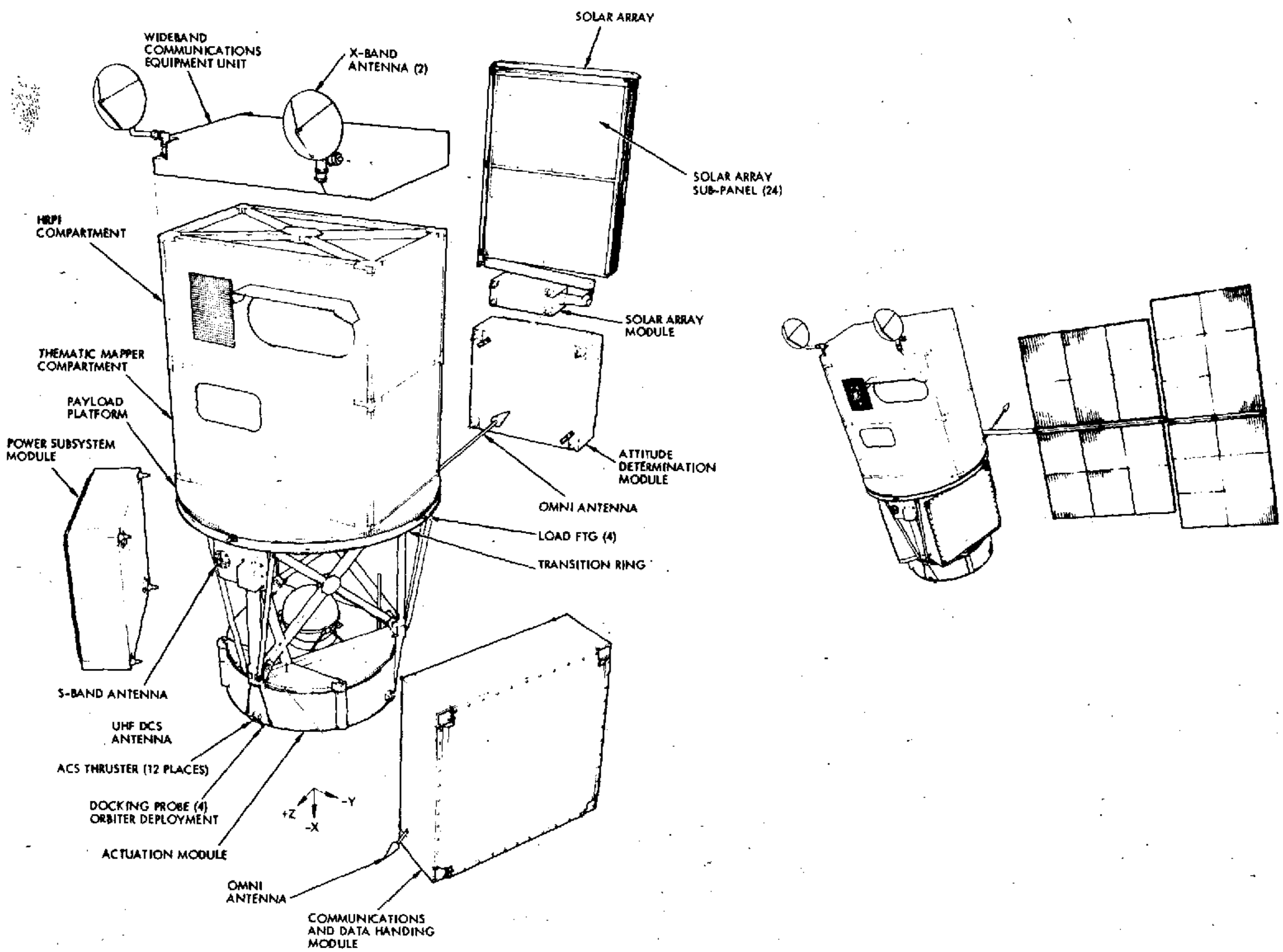

Figure 4-3. Thor-Delta Configuration for EOS-B

4.2.1 Payload Characteristics

The EOS-B mission uses the same thematic mapper as EOS-A.

The baseline HRPI instrument is a solid-state multispectral imager. A linear array of 4864 photodiodes in each of four color bands collects image data in a pushbroom scanning mode. Based on the Westinghouse HRPI configuration, the sensor provides ±30-degree cross-track field-of-view pointing and 10-meter ground resolution. The Westinghouse pointing mirror and telescope have been reoriented to obtain nonskewed off-nadir imagery.

The solid-state design has inherent high geometric registration and mechanical reliability. Since image scanning requires no moving parts, vibrational impact on other satellite subsystems is minimal. Detector array sensors offer weight and/or signal-to-noise advantages over corresponding mechanical scanners. Increased detector dwell time results in higher signal-to-noise.

The baseline HRPI is configured for a 705-km orbit altitude. Using lightweight materials (beryllium optics and structure) and present detector technology, the baseline instrument weighs about 300 pounds.

The array for each color has 4864 detectors (19 chips with 256 detectors per chip). Associated with each detector chip are four analog multiplexers (Figure 4-4). Each multiplexer sequentially samples 64 detectors. This results in a total output of 304 parallel analog data lines. The multimegabit operation data system (MODS), packaged as part of the HRPI electronics, converts the analog signals to a single line of digital data. Interlaced with the digital data are synchronization and housekeeping data.

The MODS control unit receives a timing signal from a spacecraft central control and timing unit. This synchronizes the HRPI and thematic mapper data rates for instrument signal combining (quadriphasing) prior to transmission. The multiplexer address decoder receives the control and timing commands from the MODS control unit and simultaneously controls the 304 analog multiplexers via a 64-element multiplexer control bus.

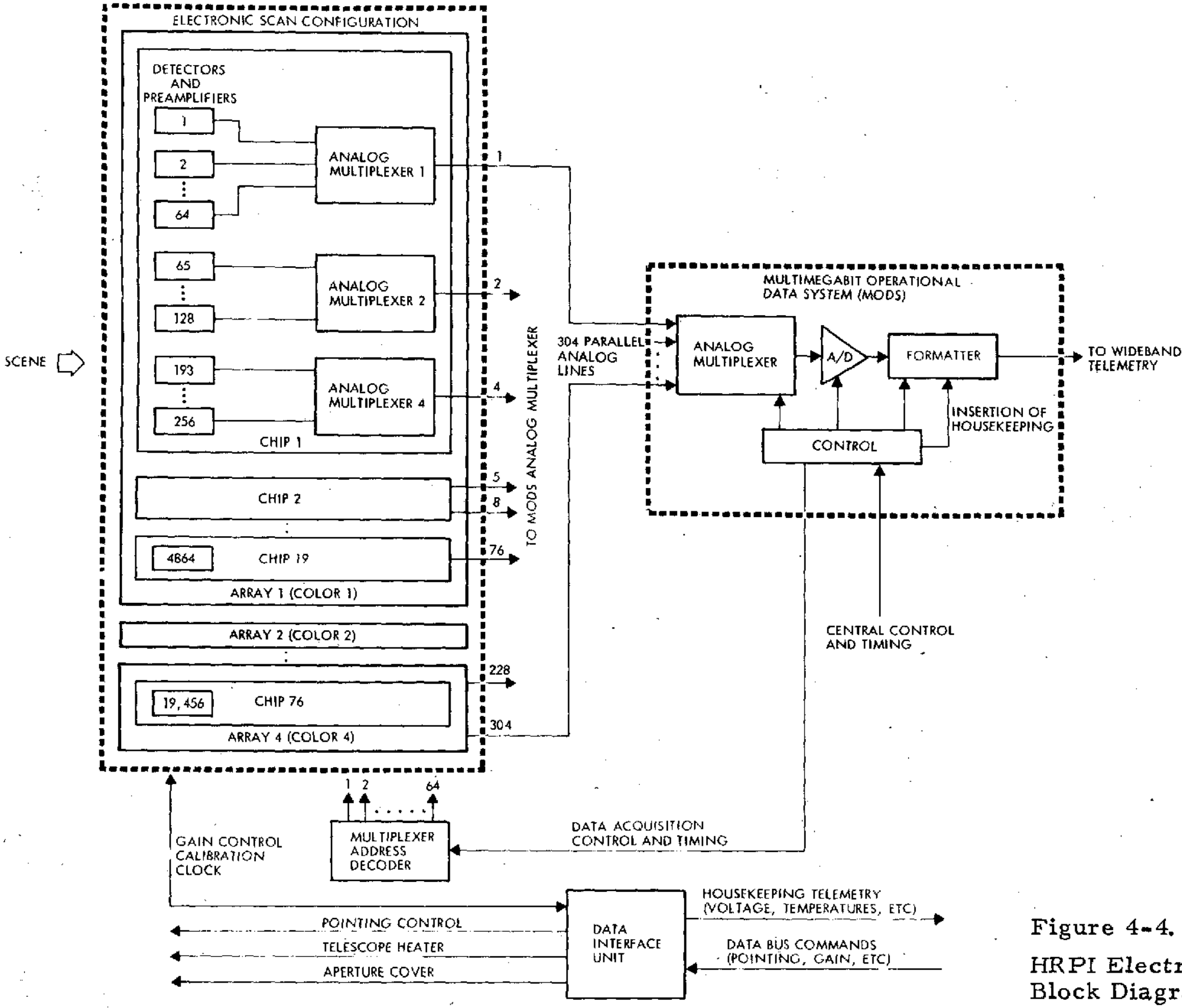

Figure 4-4.

HRPI Electronics Block Diagram

Data bus commands (mirror pointing, electronic gain, calibration, etc.) are received by the data interface unit and translated for instrument function control. HRPI housekeeping information (voltages, temperatures, electronic gain state, etc.) is formatted by the DIU for housekeeping telemetry.

4.2.2 On-Board Payload Data Handling

Data handling for the thematic mapper in both its full capacity and compacted form duplicates EOS-A. A second MODS unit running synchronously with that for the TM is required for the HRPI. Because of the order in which data are gathered, programming the MODS controller for this unit requires a substantially different PROM than for the TM. This, of course, is provided for in the MODS design.

The 120-Mbit data stream emerging from the second MODS is combined with the data stream from the thematic mapper in a quadriphase modulator, just as the two streams are merged for EOS-A. From this point, there are no differences in the data handling until the data arrive at the buffer/reformatter at the CDPF.

4.2.3 Standard Spacecraft Bus

The selection of mission-peculiar options for EOS-B is essentially the same as for EOS-A. Principal differences result from the inclusion of an orbit transfer capability and on-orbit serviceability. These in turn require increases in the nitrogen and hydrazine carried and the inclusion of a 50-pound hydrazine thruster.

4.2.4 Ground Data Handling

A three-computer configuration is required to handle the throughput of the EOS-B mission, which includes two 120 Mbit instruments, and to generate all the desired user products. By postulating that EROS will supply all user products, the system can drop to two computers.

The buffer/reformatter interface between the HDMR and the computer is different for the HRPI than for the TM. This, however, is completely within the scope of the unit design and its impact is minimal. Certain software routines are different for the two instruments, but this is anticipated in designing for modularity.

4.3 SOLAR MAXIMUM MISSION (SMM)

In 1979, the sun's flare activity will reach a maximum and provide an opportunity for scientific data gathering that occurs about once every 11 years. To capitalize on this opportunity, the solar maximum mission must be launched in 1979. Unlike EOS-A and -B, SMM is solar rather than earth pointing. Figure 4-5 shows the solar maximum payload attached to the standard spacecraft bus; Tables 4-1 and 4-2 specify standard bus mission peculiars and a weight budget, respectively. Synthesizing a SMM observatory from standard modules would be a thorough test of the flexibility and low cost of this design approach.

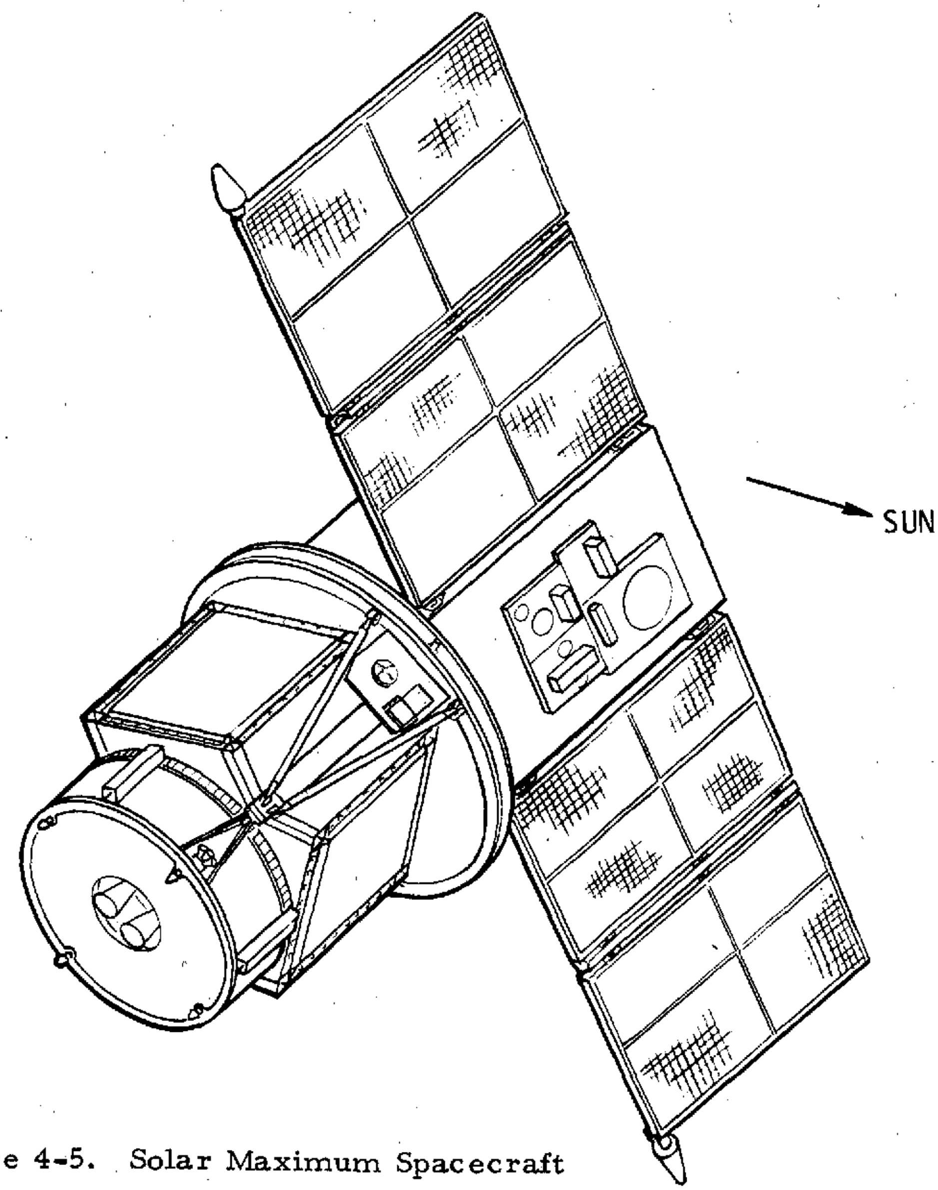

Figure 4-5. Solar Maximum Spacecraft

4.3.1 Solar Maximum Mission Payload

The SMM will investigate the cause and nature of solar flares during the period of maximum solar activity. A tentative payload consists of the 14 instruments listed in Table 4-5. These represent the instrument complement from the OSO-K Solar Flare Mission Study, plus a coronagraph.

The total data bandwidth for continuous operation of all instruments is about 6.4 kbit/sec. The whole complement of instruments will generally be pointed at an active region on the sun, with a pointing accuracy of 1 to 5 arc-sec RMS in pitch and yaw, with knowledge after the fact to 5 arc-sec. Roll accuracy and knowledge are 6 arc-sec. Stability of 1 arc-sec RMS in pitch and yaw is required for a 5-minute duration.

Total payload weight is 1100 pounds and power consumption is 200 watts. Instruments will normally operate continuously. The most likely orbit will be from 28 to 33 degrees inclination at an altitude of 500 to 550 km.

4.3.2 On-Board Payload Data Handling

Data rates for the SMM payload are sufficiently low that all data can be transmitted on the unified S-band subcarrier allocated for low to medium rate data. The only payload-peculiar equipment required is a multiple-speed tape recorder, which is mounted inside the CDH module. For this mission, data are gathered from the instruments using the data bus, thus providing complete format flexibility. For real-time transmission, gathered data are delivered to the subcarrier modulator. During storage for later playback, gathered data are delivered to the tape recorder.

4.3.3 Standard Spacecraft Bus

Drag makeup is unnecessary for the solar maximum mission; thus, the most striking alteration of the standard bus for SMM is the deletion of the hydrazine propulsion system. This is also compatible with a Delta launch, which can inject directly into orbit, and with the relatively short mission life which obviates on-orbit resupply. Should retrieval by the Shuttle be desired, rendezvous has to take place at the observatory altitude which is well within the Shuttle capability.

Table 4-5. Typical Instrument Complement for MSS

Instrument	Spectral Range	Maximum $E/\Delta E$ or $\lambda/\Delta\lambda$	Spatial Resolution (arc sec)	Field of View (arc min)*	Time for Significant Measurement (sec)
UV magnetograph	1100 to 1200 Å	10^5	2	-	1
EUV spectrometer	20 to 700 Å	3×10^4	2	-	0.5
High-resolution X-ray spectrometer	1 to 25 Å	10^4	5	-	0.1
Hard X-ray imaging	10 to 80 keV	10	4	30 x 30	30
Low-energy X-ray polarimeter	1 to 20 Å	10^3	10	-	3
Medium-energy X-ray polarimeter	10 to 50 keV	10	-	-	10
Gamma ray detector	300 keV to 10 meV	20	-	-	60
Hard X-ray spectrometer	10 to 500 keV	5	-	-	0.1
Solid-state X-ray detector	1 to 20 keV	50	-	-	0.3
Coronagraph	White light	-	2	30 x 30	10
UV spectrometer	500 to 1500 Å	10^4	2	-	10
Neutron detector	10 to 100 meV	5	-	-	60
H-α photometer	6563 Å	10^4	2	5 x 5	0.1
Flare finder	1 to 15 Å	5	10	30 x 30	0.1

*If different from spatial resolution.

The communication and data handling module retains its baseline configuration except for the addition of a tape recorder discussed in Section 4.3.2. Additionally, the bus controller interfaces with more data interface units because there are more instruments and one is required for each. This is a software rather than hardware consideration, however.

The electric power module is unchanged from the baseline (two 40 amp-hr batteries). The solar array is fixed on the observatory body after deployment and uses only 16 of the standard subpanels.

Attitude control accuracy has to be improved somewhat. This is easily accomplished by providing a more accurate attitude reference. This comes from the sun itself through a flare location unit, which is part of the payload.

4.3.4 Ground Data Handling

Because of the low data rates for the MSS, the modular ground data handling system is inappropriate. Instead, STADAC and Telops become natural selections. Thus, principal investigators receive their data in the normal way as for any other scientific experiment.

5. VERIFICATION CONCEPT

Our verification approach is an innovative but sound departure from the conventional. It reduces test program costs substantially yet achieves the required confidence levels, and is compatible with the modular implementations of spacecraft equipment. In general, the recommended approach calls for extensive qualification testing to validate design parameters of the modules, and limited acceptance testing at the spacecraft and observatory level. Table 5-1 is a comparison of the recommended versus conventional test programs.

Table 5-1. Overview of Recommended Versus Conventional Test Programs

	Recommended	Conventional
Qualification		
Component (unit)	Functional Random vibration (3 axes) Thermal vacuum EMI/EMS (1)	Functional Sine and random vibration (3 axes) Thermal vacuum or solar simulation Shock EMI/EMS
Subsystem (module)	Functional Acoustics Thermal vacuum EMI/EMS	Functional (2) Thermal (2)
Spacecraft	Payload interface checks	Functional Payload interface checks
Observatory	Functional Low frequency sine vibration (3 axes) Acoustics Ordnance firing shock Thermal vacuum Electromagnetic compatibility	Functional Sine and random vibration (3 axes) Modal survey Acoustics Ordnance firing shock Solar simulation and/or thermal vacuum Electromagentic compatibility Static load
Acceptance		
Component (unit)	Functional Random vibration (3 axes) Thermal vacuum	Functional Sine and random vibration (3 axes) Thermal vacuum
Subsystem (module)	Functional Acoustics (4) Burn-in (4)	Functional
Spacecraft	None	Functional Payload interface tests
Observatory	Functional (5) Acoustics (5) Burn-in (5)	Functional Acoustics or vibration Thermal vacuum Burn-in (6)

NOTES:
(1) Engineering data only
(2) Large integrated payloads only
(3) Limited
(4) Refurbish items only
(5) Complete observatory launch
(6) Special cases (i. e., storage, launch delay, or suspected infant failure)

The module designs provide a minimum number of standard interfaces and the isolation required to prevent interaction at the system level. Once these interfaces are verified and margins determined, the system-level test program can be reduced effectively without compromising reliability. For our verification approach, therefore, the critical test is interface verification. Of course, internal functional performance determination is important, but conventional designs and existing hardware are used for these components. The major advance is in design and control of the module interfaces; therefore, the major thrust of the test program is in that direction.

Our verification approach takes into account the following factors:

- The test program must provide as a minimum the same level of confidence achieved on conventional test programs.
- There are no differences in design between qualification and operational modules.
- Both the qualification and operational modules are fabricated by the same manufacturer with the same processes and inspection methods.

The conventional test plan emphasizes testing at both the component (unit) and all-up observatory levels, with limited (or no) testing at the module level.

If we assume that the same degree of testing is required to achieve the same level of confidence, the number of environmental test hours is lowered on the order of 50 percent. The ability to take advantage of the reduced test time depends upon developing adequate test methods for verification of module interfaces (electrical, mechanical and thermal), and reducing the number of failures and retest at the module level. The key to reducing observatory-level tests lies in the module interface verifications. The key to qualification and acceptance testing at the module level lies in reducing component failures that require module repair and retest.

Having the integrating contractor perform both module and observatory integration and testing is efficient and cost-effective. The following factors support this approach:

- Redundant testing between the module and observatory levels is minimized.

- One set of EGSE and MGSE supports both module and observatory test operations. Common equipment is shared between modules to reduce the total quantity needed for the overall project.

- Start-up and familiarization are virtually eliminated for the observatory integration and test operations.

- Module and observatory functional performance tests are similar so that one set of plans and procedures can be used both both.

Figures 5-1 and 5-2 depict the low cost integration and test flows for the baseline spacecraft. For flight acceptance, the conventional thermal vacuum test is deleted in favor of a burn-in test that has been used effectively on two military projects at TRW Systems.

5.1 QUALIFICATION

One component of each type, each module, and the all-up observatory are subjected to the qualification environmental and performance tests given in Table 5-1. The emphasis is on characterizing the performance of each module interface - thermal, mechanical and electrical. Our recommendations for qualification testing are as follows:

- Shock testing should be limited to an ordnance firing simulation with a complete observatory.

- Acceleration tests are not required.

- Electromagnetic compatibility (EMC) tests at the observatory level should be limited to noise measurements at the module test connectors. Coupled with an integrated system test, these measurements should reveal all mutual electromagnetic compatibility problems.

- No spacecraft electrical bench tests separate from the observatory should be required.

- Mass properties can be limited to measurement of weight and center of gravity.

- No static tests are required because of the increased factor of safety for the structure.

5.2 ACCEPTANCE

Each component, each module, and the observatory are subjected to the flight environmental and performance tests given in Table 5-1.

Our recommendations for acceptance testing are as follows:

- An electromagnetic interference and susceptibility test for modules is not warranted for workmanship verification.
- A separate spacecraft-level pre-integration functional test is not required and is not cost-effective.
- A burn-in test should be performed at the module level for refurbish items and at the all-up level for a launch of a complete observatory. This serves as a low-cost thermal test and detects infant failures.

5.3 DESIGNING FOR LOW-COST TESTS

Certain design characteristics that have cost impact upon the system test program are:

- Testability. Tests can be simplified or conducted more efficiently if testability features are incorporated in the design.
- Accessibility. Observatory integration and failure troubleshooting can be done more simply and rapidly if easy access if provided to modules, components, and cable harness connectors.
- Design margins. Tests can be reduced or even eliminated if the design margin exceeds the value considered safe.

5.3.1 Testability

Testability provisions are required primarily to simulate and measure response of the observatory. These features are provided via the command/telemetry links and/or hardline test points. The ability to program the on-board computer for any desired command sequence and telemetry format greatly simplifies hardline requirements.

For system test operations, the spacecraft telemetry and commands are used for monitoring and stimulus. Module test point hardlines are required to verify observatory performance, to isolate failures to a replaceable component, and to expedite test operations. These hardlines provide the following:

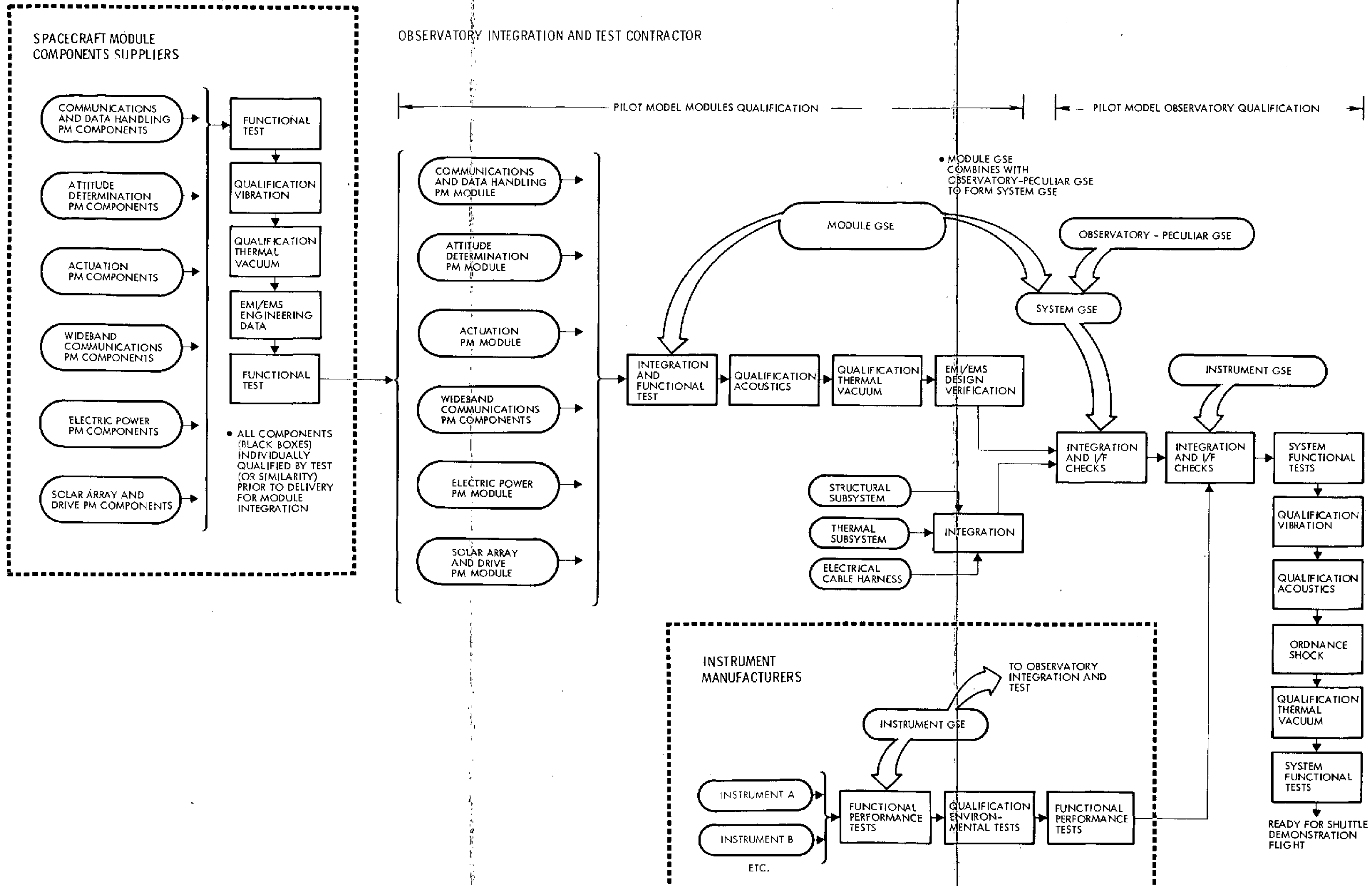

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Figure 5-1. Qualification Model Integration and Test Flow

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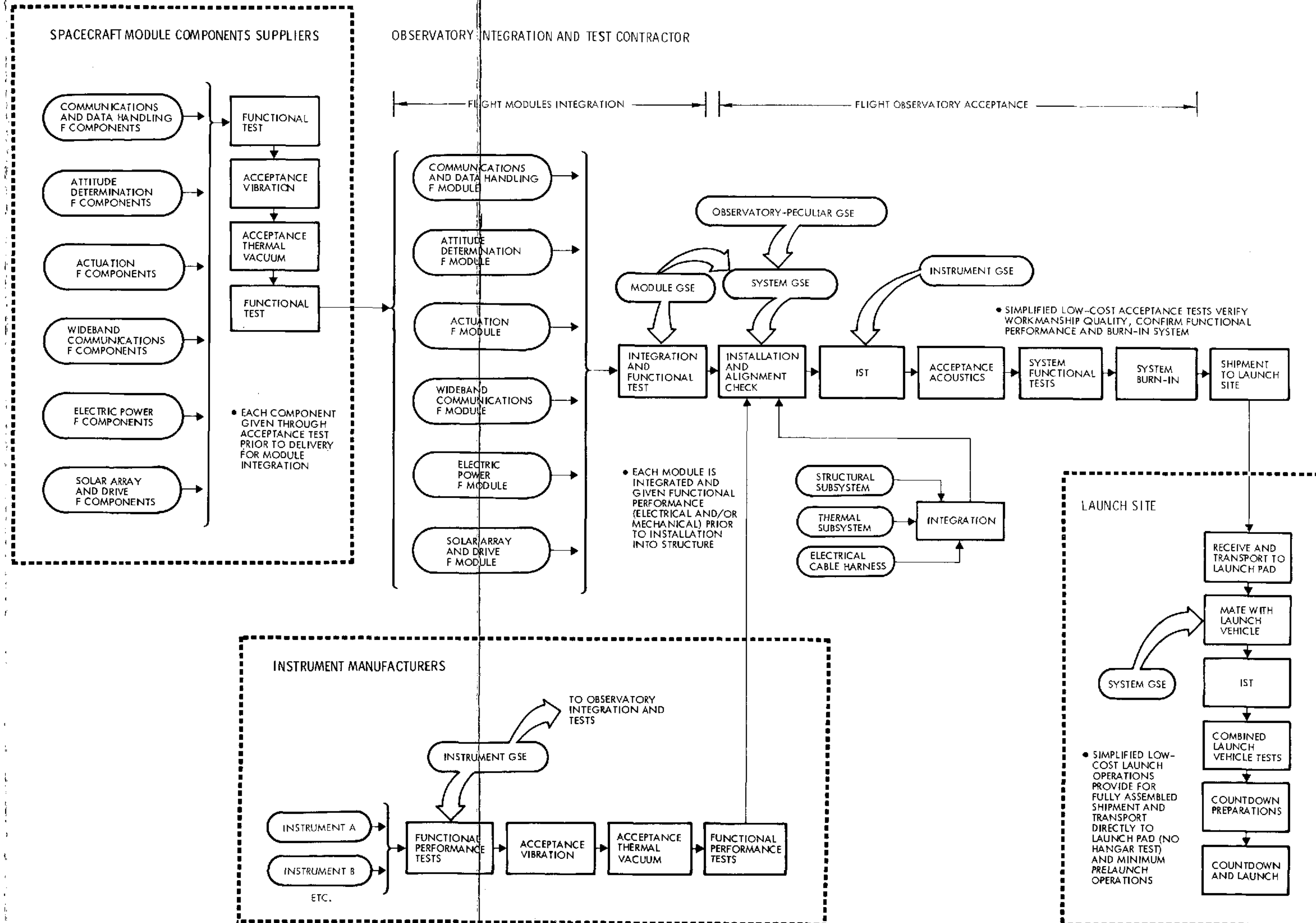

Figure 5-2. Flight Integration, Test and Launch Operations Flow

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FOLDOUT FRAME 2

- Supplement telemetry
- Supplement command inputs
- Simulate sensor outputs when physical stimulation is not feasible or cannot produce the output characteristics required
- Prevent breaking electrician interconnections for test
- Provide EMI measurements

If the number of test points becomes excessive in terms of space, weight, or cost, the priority for their incorporation to facilitate testing is system performance verification, redundancy testing, and failure isolation to a replaceable component.

Testability provisions also arise out of requirements to service the modules and observatory both in-plant and at the launch site.

Specific testability features incorporated into the module design are as follows:

- Test connectors on each module are accessible after complete assembly of the observatory.
- All test points are isolated to preclude damage due to grounding the signal. The isolation circuit accommodates the EAGE load/source impedance including cable resistance/capacitance.
- All test connectors are female.

Specific testability provisions incorporated to facilitate testing the complete observatory after mating with the launch vehicle are as follows:

- Command input via both the module test connector and the umbilical which does not require acquisition of the RF uplink.
- Telemetry output via the module test connector and the umbilical which does not require an acquisition of the RF downlink.
- External power input to charge each battery at maximum and trickle charge rates with the observatory de-energized; battery temperature monitoring via hardline test points.
- Deep discharge of each battery for reconditioning with the observatory de-energized.

- Control and monitoring of the actuation valves with the observatory de-energized; open/close control of each valve, open/close status of each valve, and tank temperature monitor.
- Measurement of RF power output from each transmitter. Monitor point as near as possible to the antenna input.
- Simulation of each AVCS sensor output and monitoring each thruster firing signal.
- Simulation of the launch vehicle-observatory separation switch actuation.
- External power input to simulate the solar array output, both via the module test connector and the umbilical.
- Verification of parallel redundant elements.
- Propellant and pressurant loading provisions.
- Ordnance bridge wire resistance and firing current measurements.
- Direct memory access to the on-board computer for loading and readout.

5.3.2 Accessibility

Accessibility has a significant impact upon integration and test costs. If modules can be installed in any order, schedule delays are reduced and integration activities can be much more flexible.

The following accessibility provisions are included in the design:

- A break-out T-connector is inserted between the module and cable harness connectors for interface signal measurements.
- Disassembly of the module structure and thermal insulation to replace a faulty component does not require retest at the module level.
- Modules and components can be installed or removed without removing other equipment.

5.3.3 Design Margins

The following structural design margins permit elimination of static tests:

Limit load	Maximum expected flight load
Design yield	1.5 x limit
Design ultimate	1.88 x limit

6. LOW COST MANAGEMENT APPROACHES

The modular observatory, assembled from more or less standard modules and mission-peculiar instruments, promises significant cost savings. In this era of substantially static space budgets, it is only with such cost-saving programs that we can increase the total number of space flights. Space technology has matured so that we need no longer ask of spacecraft subsystems, "can it be done?" but only, "for how little can we do it?" Now we need to match that technical maturity with innovative management approaches, approaches that enhance technical cost savings, approaches that lead to predictably low costs. We need a "modular" management system.

Basically the modular management system partitions management tasks much as the modular spacecraft partitions system functions. It too relies on realistic and compatible interfaces where required information exchanges are minimized and well defined. In essence we believe it possible to manage a large and complex program as efficiently as a tightly knit project-oriented small program. In fact, the management overhead structure would be little larger than the sum of the managements for the smaller segments rather than the substantial pyramid common to conventional programs.

This efficiency is possible if we carry the modular technical concept to its logical conclusion. If early in the program we specify realistic and compatible technical interfaces between modules and if throughout the program we rigidly enforce them without deviations, the need for multidisciplinary communication across segment lines largely disappears. Thus, after agreement is reached in the planning phase, a module manager need not be concerned with compatibility with segments outside his control but only with whether or not he is meeting his own performance and interface specifications.

Through this modular management approach, controls and customer reporting requirements can be tailored to match each segment's specific needs. Resources and schedules can be planned and managed as best suited to each manager's needs. A sense of teamwork enhanced by mild competitive feelings between module teams will add to efficiency previously unobtainable on large programs.

One way the modular management approach keeps activities manageable is by temporarily separating planning from performing. Early in the effort the program manager leads a small task force of key people in analyzing and specifying every module interface, technical performance requirements, schedule milestone and target cost. This task force is composed of module managers and key system specialists. All are veterans of prior space programs to ensure reality in the evolving system definition and to avoid the all too common pitfall on spacecraft programs of performance specifications that push the state of the art, unbeknown to either the government or the contractor. Each module manager will be asked to commit himself to meeting the goals and specifications set for his module.

We expect the planning phase for EOS to take from 3 to 6 months. When the planning phase is complete and commitments have been obtained from all module managers, the modular management approach enters its second phase; one dominated by delegation. Each module manager carries on as the head of a nearly autonomous organization, with the advantages of a small close-knit group retained, though the number of groups has proliferated. Each module manager, by virtue of his role in the planning phase, knows what is expected of him and why. More of his attention is directed inward at his own module; he has the time to identify and correct problems while they are still embryonic, and the incentive of knowing he has no place to pass the buck.

The program manager in the second phase does not require the traditional massive staff of a large program. His main attention is on technical progress and how it relates to schedule and cost. He is the formal point of contact with the customer, while ensuring that informal contact continues daily at other levels within the program. He retains control of such program-wide efforts as parts standardization, formal reporting, system engineering, and cost reporting. The months of planning have given him a clear picture of exactly how the program should progress, and deviations are immediately visible. Moreover, he has heard all the arguments for and against each specification imposed, has worked to get the agreement of each module manager to his specifications, and is not likely to change his policy of no deviations to specifications.

We estimate that millions of program dollars, 10 to 15 percent of a program's cost, can be saved through implementation of a modular management approach. In summary, we make the following specific recommendations for achieving a low-cost EOS program:

- Manage for low cost with a modular management approach decentralized to place on each module manager the unambiguous responsibility for meeting performance and interface specifications within cost and schedule.

- Ensure close interaction with NASA personnel by collocating the NASA people with the program to provide the contractor with rapid guidance when unanticipated decisions are required, and, with a minimum of expensive formal reporting, provide assurance to the government that all is well.

- Use the contractor's existing program management systems. Coupled with an earned value capability (for progress management) and a design-to-cost system (for recurring cost measurement), the contractor's internal system can provide the basic data needed by contractor program management to control costs and schedule performance and to report progress to NASA.

- Reduce manpower costs by limiting formal design reviews to the module level and higher, and holding only informal reviews at the black box level. At formal reviews, customer participation is at its peak and the contractor responds with an excessive expenditure of manpower/costs. Requiring only informal working reviews at the black box level minimizes this type of response.

- For formal configuration control with the customer, limit control documents to the statement of work, system specification, and interface control specifications. Release black box specifications after the critical design review and allow control by the subsystem manager.

- Use a combination CPIF/CPAF contract for the systems integrator. Use CPIF incentives on cost and measurable parameters, and CPAF incentives on non-quantifiable parameters, such as management performance and payload interface management.

- Eliminate the "contract changes" clause and negotiate every departure from the original agreement by bilateral action (supplemental agreement). This eliminates questionable changes and decreases the cost of contract change administration.

- Provide zero fee for all contractor-initiated changes and double the fee rate for all NASA-directed changes. This discourages change activity on the part of both parties.

- Share all program cost savings originated by the contractor on a predetermined percentage basis to motivate the contractor to cut costs.

- Implement a formal contractor program for cost awareness and personnel motivation.